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GOODYEAR

GOODYEAR AIRCRAFT CORPORATION

AKRON, OHIO

STRESS ANALYSIS OF INFLATOPLANE

MODEL GA-468

GER-9861
897-3

Contract No.
NOnr 2368(00)

January 10, 1961

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REVISIONS

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REFERENCES

1. MIL-HDBK-5, Strength of Metal Aircraft Elements, March 1959.

2. Goodyear Aircraft Corporation, Structures Design Manual.

3. GER-8860, Effect of Cross-Section Shape on Stress Distribution in Inflated Airmat Panels, dated June 10, 1958.

4. GER-9742, Summary Aerodynamics Report of One-Man Inflatoplane, Model GA-468, dated 10 August 1960.

5. GER-10012, Preliminary Results of the Wind-tunnel Investigation of the Aerodynamic and Structural Deflection Characteristics of Model GA-468 Inflatoplane, dated 14 October 1960.

6. Timoshenko and Woinowsky-Krieger, Theory of Plates and Shells, Second Edition, 1959, McGraw-Hill Book Company, Inc., New York.

7. Civil Aeronautics Manual 3, "Airplane Worthiness; Normal, Utility, and Acrobatic Categories." Spetember 1954.

8. GER-7759, "Stress Concentrations and Deflections in Axi-symmetrically Loaded Spherical Envelopes", by A. D. Topping and J. D. Marketos, February 8, 1957.

9. MIL-A-8629, Airplane Strength and Rigidity, 20 August, 1953.

REFERENCE DRAWINGS

447A-001	Assembly One Place Inflatoplane
447A-002	Wing-Assembly of
447A-003	Fuselage - Assembly of
447A-004	Empennage - Assembly of
447A-005	Cockpit - Assembly of
447A-007	Engine and Mount - Assembly of
447A-011	Engine Mount
447A-013	Universal Gear - Assembly of
447A-025	Bonded Assembly One Place Inflatoplane

INTRODUCTION

This report is submitted in partial fulfillment of paragraph 10 of Amendment 8 to Office of Naval Research Contract NOnr 2368(00), "Inflateplanes".

The Stress Analysis for GA-468 Model Inflateplane is divided into six sections. These sections are listed below.

Section 1	General
Section 2	Load Analysis
Section 3	Wing Analysis
Section 4	Fuselage Analysis
Section 5	Engine Mount Analysis
Section 6	Summary of Landing Gear, Cockpit, and Empennage Analyses

DISCUSSION

The Inflatoplane is a high wing monoplane with a Nelson H63A engine mounted on a pedestal above the wing and fuselage near the trailing edge of the wing. Each wing panel is restrained by two guy cables on the upper surface and three on the lower surface. The two upper cables are anchored to the engine pylon. The forward outboard and inboard cables are tied to the landing gear, while the aft outboard cable is attached to the bottom of fuselage at fuselage station 105.00. A single-seat cockpit is located forward of the conical shaped fuselage. A single wheel landing gear is mounted on the front hemi-spherical end of the fuselage; this together with the wing tip skids and a tail skid make up the landing gear system. Conventional tail surfaces are restrained by guy cables attached to the fuselage. A general layout with pertinent geometric data is shown in Figure 1.

The GA-468 Inflatoplane was static tested at Goodyear Aircraft to determine the buckling strength of the wing. An ultimate load factor of 5.6 was obtained.

A wind tunnel test was conducted in the NASA Langley full-scale wind tunnel to corroborate the static test results. This investigation resulted in ultimate load factors up to 5.13, reference 5.

An endurance load test was conducted at Goodyear Aircraft to determine the time effect under limit load on the Inflatoplane fabric. The Inflatoplane was inverted and a 2.5g limit load was applied to the wing and fuselage using shot bags. The test was run for 336 hours without appreciable creep.

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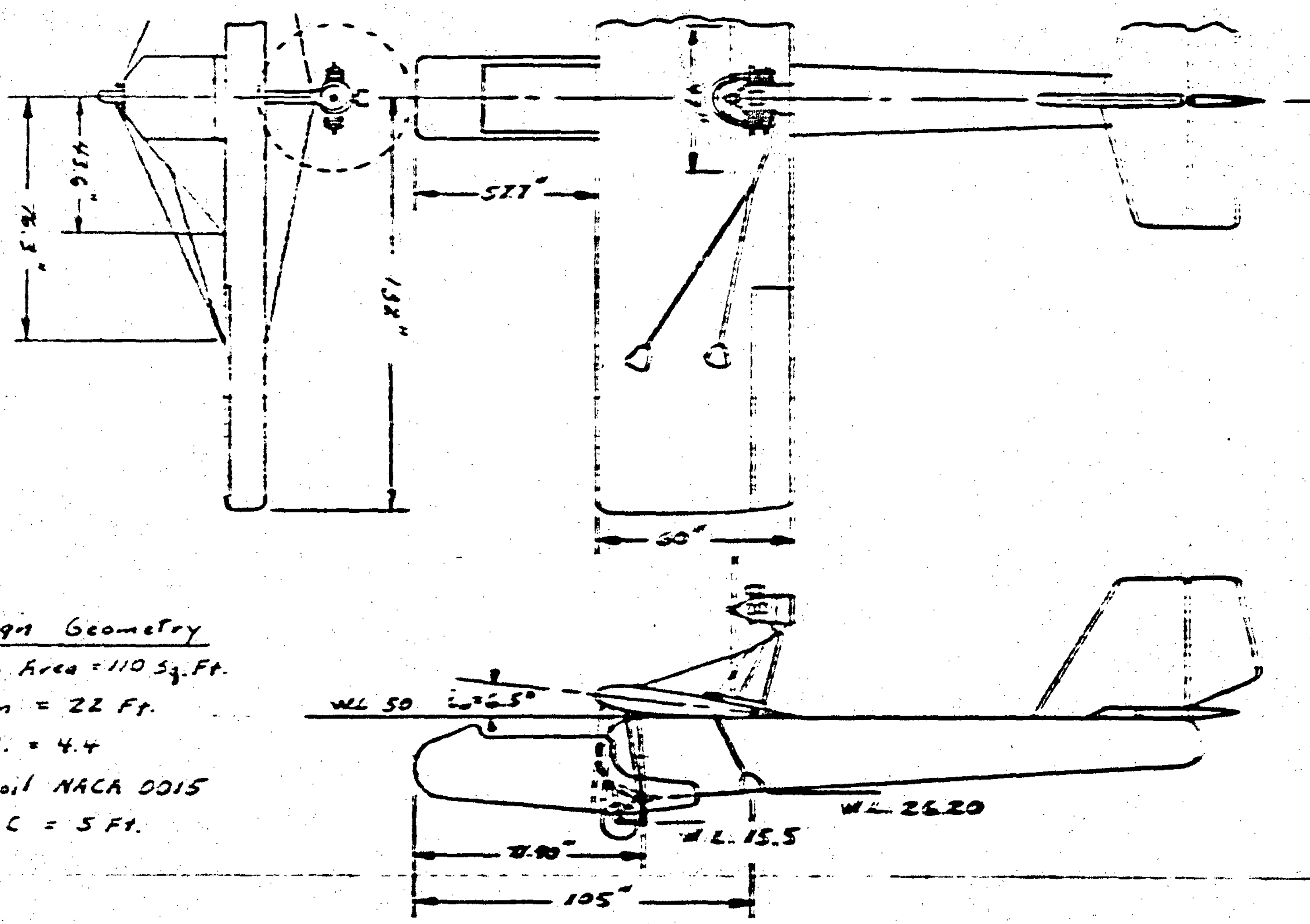

Design Geometry

Wing Area = 110 Sq. Ft.
Span = 22 Ft.
A.R. = 4.4
Airfoil NACA 0015
MAC = 5 Ft.

Figure 1 Geometric Characteristics Of The Inflatoplane

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TABLE 1

CLOTH SPECIFICATIONS, [illegible]

1.	Goodyear Code	477	484	3303N	3511N	3314N	33[illegible]
2.	Material	Dacron	Dacron	Nylon	Nylon	Nylon	Nyl[illegible]
3.	Nominal Weight oz/sq.yd.	4.00	3.50	3.10	2.03	0.60	1.[illegible]
4.	Weight Tolerance oz/sq.yd.	± .15	± .20	.6	± .10	± .25	± .[illegible]
5.	Tensile-Min-lbs/inch-Warp	200	275	152	90	160	[illegible]
6.	-Fill	180	250	132	90	125	[illegible]
7.	Tensile Test Method (Quick Break)	1" Strip	1" Strip	1" Strip	1" Strip	1" Grab	2" [illegible]
8.	Denier-Warp	220/1	220/1	210/1	70/2	70/2	40/[illegible]
	-Fill	220/1	222/1	210/1	70/2	210/1	70/[illegible]
	-Pile	---	---	---	---	70/2	---
9.	Count-Warp, Minimum Ends/inch	62	84	49	96	73	11[illegible]
10.	Count-Fill, Minimum Ends/inch	55	78	49	96	40	[illegible]
11.	Count-Pile-Nominal, Tolerance (yarns/sq.in.)	--	--	--	--	30, ±5	-

1

TABLE I

CLOTH SPECIFICATIONS, [illegible]

404	3903N	3511N	3514N	3523N	2044	3354	7008B	8936	8937
Dacron	Nylon	Nylon	Nylon	Nylon	Dacron	Dacron	Nylon	Nylon	Nylon
5.50	3.10	2.03	0.60	1.20	6.50	5.00	4.25	15.00	9.25
± .20	.6	± .10	± .25	± .10	± .20	± .15	± .15	± .40	± .30
275	152	90	160	50	312	237	231	160	160
250	152	90	125	50	293	233	210	125	125
1" Strip	1" Strip	1" Strip	1" Grab	2" Strip	1" Strip	1" Strip	Grab	Grab	Grab
220/1	210/1	70/2	70/2	40/1	220/2	220/1	210/1	70/2	70/2
222/1	210/1	70/2	210/1	70/2	220/2	220/1	210/1	210/1	210/1
---	---	---	70/2	---	---	---	---	70/2	70/2
84	49	96	73	117	50	79	71	73	73
70	49	96	40	52	49	73	71	40	40
--	--	--	30, ±5	--	--	--	--	30, ±5	30, ±5

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TABLE II

FABRIC SPECIFICATIONS, INFLATOPLANE

Processed Fabric Specifications

		1	2	3	4	5
		(Present)	(Proposed)		Empennage Aileron	
1.	Classification	Wing	Wing	Cockpit	& Flap	Fuselage(2)
2.	Goodyear Code	A350	A350	A351	A349	H313A105
3.	Outside Color	Plain	Plain	Plain	Plain	Plain
4.	Number of Plies	3	3	2	2	2
5.	Construction (outside to inside)	(1)	(1)	(1)	(1)	
	a.) Spread (oz/sq yd)	1.25	1.35	---	---	1.20
	b.) Cloth "	1.404L	2.055L	---	---	4.008
	c.) Spread "	2.50	2.70	1.25	1.25	4.50
	d.) Cloth "	1.408R	2.055R	2.053	2.053	4.008
	e.) Spread "	3.00	3.10	5.50	5.00	1.00
	f.) Airmat Cloth "	15.003	15.003	8.608	9.258	----
	g.) Spread "	----	----	----	----	----
6.	Nominal Weight - oz/sq yd	34.00	39.50	26.20	26.00	14.70
7.	Weight Tolerance - oz/sq yd	1.70	1.95	1.25	1.50	.50
8.	Tensile-Min-lbs/inch-Warp	180	180	150	140	300
9.	-Min-lbs/inch-Fill	174	174	150	140	260
10.	Min-lbs/sq inch-Pile	28	28	28	28	---
11.	Tensile Test Method					Cyl burst
12.	Material	Nylon	Nylon	Nylon	Nylon	Dacron
13.	Cloth-Outside to Inside	3523N	3511N	3511N	3511N	
		3523N (1)	3511N (1)	3514N (1)	8937 (1)	477-477
		8936	8936			

(1) For Airmat construction each side is symmetrical.
(2) Also used for cockpit assembly straps.
(3) Used for reinforcement of fuselage fabric H313A105 on aircraft 4106-4113.
(4) Used for replacement fuselages on aircraft 4114 and 4115.
(5) To be used as replacement fuselages on aircraft 4106, 4108-4113.
(6) Used for hinge straps, D-ring, pulleys and fan patches, and reinforcement in lacing p
(7) Used for instrument brackets, lacing patches, scuff patches, and seam tape on wings.

TABLE II

FABRIC SPECIFICATIONS, LIGHTER-THAN-AIR

	3	4	5	6	7	8	9	10
		Empennage						
ed)		Aileron						(7)
	Cockpit	& Flap	Fuselage(2)	(3)	(4)	(5)	Strap (6)	Accessory
	A351	A349	N313A105	N3[illegible]A10	ZX321	ZX 356	ZX300	A330
	Plain	Plain	Plain	Plain	Plain	Plain	Plain	Plain
	2	2	2	2	2	2	2	1
	(1)	(1)						
	---	---	1.20	1.10	1.20	3.30	1.25	4.10
	---	---	4.008	4.003	3.00[illegible]	3.50[illegible]	4.233	3.10
	1.25	1.25	4.50	4.50	4.90	[illegible]	4.50	4.50
	2.033	2.033	4.003	4.003	6.303	5.303	4.233	----
	5.50	5.00	1.00	1.10	1.09	1.00	1.25	----
	8.603	9.233	----	----	----	----	----	----
	----	----	----	----	----	----	----	----
	26.20	26.00	14.70	14.70	17.70	18.00	15.50	12.1
	1.25	1.50	.50	.50	.50	.50	1.00	.5
	150	140	300	300	435	410	425	150
	150	140	260	300	425	410	350	150
	20	20	---	---	---	---	---	---
			Cyl Burst	Cyl Burst	Cyl Burst	Cyl Burst	Strip	Grab
	Nylon	Nylon	Dacron	Dacron	Dacron	Dacron	Nylon	Nylon
	3511N	3511N						
(1)	35148 (1)	0937 (1)	477-477	477-477	3364-2014	464-484	7008B-7008B	[illegible]

metrical.

o N313A105 on aircraft 4106-4113.
ft 4114 and 4115.
ircraft 4106, 4108-4113.
nd fan patches, and reinforcement in lacing patches.
ches, scuff patches, and seam tape on wings.

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Summary of Minimum Margins of Safety

Page No.	Part Name	Drawing No.	Critical Condition	Type of Loading or Stress	M.S.
3.01.210	Wing, Patches & Cables	1171-002	Wind Tunnel Test	Ultimate	+0.17
4.01.050	Fuselage	1171-003	Condition F-11 & F-13	Hoop Tension with Longitudinal Stress = 0	+0.03
5.01.030	Engine Mount -11 Strut	1171-001	Ult. Side Load	Combined Axial, Bending & Torque	+0.01
6.01.050	Empennage	1171-004	A-11 Rudder Kick	Limit Bending	+1.91
6.01.150	Cockpit	1171-005	Condition 5	Ultimate Bending	-0.02
6.01.170	Landing Gear -13 Ring & -17 Tee Brace	1171-013	Condition 3	Ultimate Bending	+0.14

Design Criteria

The Design Criteria covers the design conditions, factors of safety, and allowable stresses for the Inflatoplane.

Design Conditions

The Design Conditions were selected using reference (9) as the specification. The maneuvering design conditions are in accordance with paragraph 3.4.2 of reference (9) for the VW classification (Special Search) except that the maximum limit load factor is 2.5 instead of 3.0. Two gross weights are investigated, namely, 550 lbs with most forward c.g. and 404 lbs with the most aft c.g. Maximum flight limit speed is 74.5 knots. Under gusting conditions the limit speed is 44 to 47 knots, based on maneuvering load factor of -1.0. The critical loads are summarized on page 2.00.030.

Ground handling loads are not critical. The landing conditions are based upon a sinking speed of 5 ft/sec. The critical landing conditions are level landing, tail down landing, and side load specified in reference (7).

Factors of Safety

The limit loads are multiplied by the following factors to obtain the structural design loads.

Metal Structure	Yield	1.15
Metal Structure	Ultimate	1.50
Fabric Structure	Wrinkling*	1.00
Fabric Structure	Ultimate	1.75

* No principal stress shall be less than zero at limit load.

Allowable Strength

For the metal structures, the requirements of paragraphs 3.2.1.2 through 3.2.1.8 of reference (9) are used.

The following reduction factors shall be applied to the Quick Break Strength of the fabric structures:

Inflation Only	4
Limit Load	3
Ultimate Load	1.5

The reduction factor for inflation and limit load are based on past experience and account for the fact that fabric under load for a period of time has a reduction in strength.

The reduction factor for ultimate load was chosen arbitrarily.

LOADS

SECTION 2

The critical flight loads shown in Table III, Summary of Critical Flight Loading Conditions were developed using Table 1 (Summary of Airloads), Figure 6 (V-n_g Diagram M.F.C.G. 350 lbs), and Figure 7 (V-n_g Diagram M.A.C.G. 404 lbs) of reference (4). The letters refer to points on the V-n_g diagram and subscripts to the different conditions applicable to the particular point. The loading conditions specified produced critical design loads for all the primary structure.

The weight distribution, C.G. locations, and moment of inertia calculations are found on pages 2.01.010 through 2.01.090.

Each part of the Loads Section has its own discussion and sketches.

Prepared by: M.G.G.
Checked by:
Date: [illegible]
Revised:

Summary of Critical Loading Conditions

Table III

Description		Designation	Velocity	Angle of Attack	Aileron Deflection	Load Factor	Angular Accel. Pitch $\ddot{\theta}$	Angular Accel. Roll $\ddot{\phi}$	Angular Accel. Yaw $\ddot{\psi}$	Wing Loads Port Lift	Wing Loads Port Drag	Wing Loads Stbd. Lift	Wing Loads Stbd. Drag	Empennage Loads Horizontal Port	Empennage Loads Horizontal Stbd.	Empennage Loads Vert.	Engine Thrust
			Knots	Deg.	Deg.		sec^{-2}	sec^{-2}	sec^{-2}	LB.	LB.	LB.	LB.	LB.	LB.	LB.	LB.
Symmetrical Maneuver		A2		13.6	0	2.5	-1.83	0	0	652	79	652	79	30	30	0	0
		A3				2.5	+3.44	0	0	652	79	655	79	-37.5	-38.5	0	100
Unsymmetrical Maneuver		A4		11	20 UP 15 DN	2.0	-1.83		0	652	74	448	40	36	29	0	0
		A5			"	2.0	+3.44		0	652	74	448	40	-41	-32	0	100
Gust	Symmetrical	A6	52.5	5.5	0	1.0	0	0	0	275	20	275	20	30.5	30.5	0	100
	Unsymmetrical	A7		5.5	0	1.0	0	0	0	275	20	275	20	37	24	0	100
Steady Sideslip		A8		5.5	0	1.0	0	0	0	275	20	275	20			64	100
Rolling Pullout	Zero Slip	A9		11	25 UP 15 DN	2.0	0	0	0	652	74	448	40			18	100
	Zero Rudder	A10		11	"	2.0	0	0	2	652	74	448	40			112	100
Rudder Kick		A11	59.5	4.45	0	1.0	0	0	0	275	22	275	22			181	100
Gust - Vertical Tail		A12	52.5	5.5	0	1.0	0	0	0	275	20	275	20			[illegible]	100
Combined Loads		A13				[illegible]	0							1	-6	[illegible]	
Symmetrical Maneuver		B2		8	0	2.5	-2	0	0	688	64	688	64	23.5	23.5	0	121
		B3				2.5	2.65	0	0	688	64	688	64	-28.5	-28.5	0	121
Unsymmetrical Maneuver		B4		6.4	25 UP 15 DN	2.0	-2		0	760	66	400	36	26	19	0	121
		B5		6.4	"	2.0	2.65		0	760	66	400	36	-35	-24		121
Gust	Symmetrical	B6	62.0	3.2	0	1.0	0	0	0	275	20.5	275	20.5	40	40	0	121
	Unsymmetrical	B7		3.2	0	1.0	0	0	0	275	20.5	275	20.5	42	32	0	121
Steady Sideslip		B8		3.2	0	1.0	0	0	0	275	20.5	275	20.5			53	121
Rolling Pullout	Zero Slip	B9		6.4	25 UP 15 DN	2.0	0									16	121
	Zero Rudder	B10		6.4	"	2.0	0									112	
Rudder Kick		B11	72.5				Same			As	Designation			A11			
Gust - Vertical Tail		B12	62.0				Speed			Restricted	in	Gusts					
Combined Loads		B13				[illegible]								1	-17.0	[illegible]	
Symmetrical Maneuver		C2		3.4	0	2.5	-1	0	0	688	53	688	53	16	16	0	140
		C3		3.4	0	2.5	1.52	0	0	688	53	688	53	-20	-20	0	140
Unsymmetrical Maneuver		C4		2.7	25 UP 15 DN	2.0	-1		0	768	59	332	32	19	13		140
		C5		2.7	"	2.0	1.02		0	768	59	332	32	-24	-16		140
Gust	Symmetrical	C6	74.5	1.4	0	1.0	0	0	0	275	23	275	23	35	35		142
	Unsymmetrical	C7		1.4	0	1.0	0	0	0	275	23	275	23	42	28		140
Steady Sideslip		C8		1.4	0	1.0	0	0	0	275	23	275	23			130	140
Rolling Pullout	Zero Slip	C9		2.7	25 UP 15 DN	2.0	0									18	140
	Zero Rudder	C10		2.7	"	2.0	0									112	140
		C11	95				Same	As	Designation	A11							

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Unsymmetrical	B7		3.2	0	1.0	0	0	0	275	20.5	275	20.5	42	32	0	121
Steady Sideslip	B8		3.2	0	1.0	0	0	0	275	20.5	275	20.5			53	121
Rolling Pullout – Zero Slip	B9		6.4	25 up 15 dn	2.0	0									18	121
Rolling Pullout – Zero Rudder	B10	V	6.4	–	2.0	0									112	
Rudder Kick	B11	97.5	Same as designation A11													
Gust – Vertical Tail	B12	120	Speed restricted in gusts													
Combined Loads	B13				B8-B12								1	-17.2	B8-B12	
Symmetrical Maneuver	C2		3.4	0	2.5	-1	0	0	682	53	682	53	16	16	0	140
	C3		3.4	0	2.5	1.52	0	0	682	53	682	53	-20	-20	0	140
Unsymmetrical Maneuver	C4		2.7	25 up 15 dn	2.0	-1		0	768	54	332	32	14	13		140
	C5		2.7	–	2.0	1.52		0	768	54	332	32	-24	-16		140
Gust – Symmetrical	C6	74.5	1.4	0	1.0	0	0	0	275	23	275	23	35	35		140
Gust – Unsymmetrical	C7		1.4	0	1.0	0	0	0	275	23	275	23	42	28		140
Steady Sideslip	C8		1.4	0	1.0	0	0	0	275	23	275	23			130	140
Rolling Pullout – Zero Slip	C9		2.7	25 up 15 dn	2.0	0									18	140
Rolling Pullout – Zero Rudder	C10		2.7	–	2.0	0									112	140
Rudder Kick	C11	97.5	Same as designation A11													
Gust – Vertical Tail	C12	74.5	Speed restricted in gusts													
Combined Loads	C13				C8-C12								1	-9	C8-C12	
Symmetrical Maneuver	E2		-3.2	0	-1.0	-3.53	0	0	-275	20	-275	20	32.5	32.5		121
	E3		-3.2	0	-1.0	2.0	0	0	-275	20	-275	20	-32.5	-32.5		121
Unsymmetrical Maneuver	E4		0	25 up 15 dn	0	-3.53		0					46	31		121
	E5		0	–	0	2.00		0					-44	-29		121
Gust – Symmetrical	E6	62	3.2	0	1.0	0	0	0	275	20.5	275	20.5	-46.5	-46.5		121
Gust – Unsymmetrical	E7		3.2	0	1.0	0	0	0					-55	-36		121
Steady Side Slip	E8		3.2	0	1.0	0	0	0	275	20.5	275	20.5			43	121
Rolling Pullout – Zero Slip	E9		0	25 up 15 dn	0	0									0	121
Rolling Pullout – Zero Rudder	E10		0	–	0	0									0	121
Rudder Kick	E11	97.5	Same as designation A11													
Gust – Vertical Tail	E12	62	Speed restricted in gusts													
Combined Loads	E13				E8-E12								-5	-9	E8-E12	
Symmetrical Maneuver	F2		-17.6	0	-1.0	-2.0	0	0	-275	31	-275	31	57	57		
	F3		-17.6	0	-1.0	2.0	0	0	-275	31	-275	31	-57	-57		
Unsymmetrical Maneuver	F4		0	25 up 15 dn	0	-2.0							46	46		
	F5		0	–	0	10.0							-46	-46		
Gust – Symmetrical	F6	34	17.6	0	1.0	0	0	0	275	31	275	31	35	35		
Gust – Unsymmetrical	F7		17.6	0	1.0	0	0	0	275	31	275	31	-40	-28		
Steady Sideslip	F8		17.6		1.0	0	0	0	275	31	275	31			28	
Rolling Pullout – Zero Slip	F9															
Rolling Pullout – Zero Rudder	F10															
Rudder Kick	F11	97.5	Same as designation A11													
Gust – Vertical Tail	F12	34	Speed restricted in gusts													
Combined Loads	F13				F8-F12								-5	-39	F8-F12	

Discussion

Weight and Balance Section

This section of this report presents the weight and balance data and the moments of inertia of the aircraft.

There are two parts. The first is the Group Weight Statement (AN-9103-D) and the second portion incorporates the moments of inertia.

In the Group Weight Statement the weight of the pilot is assumed to be 200 lbs. For the most forward C.G. condition, 240 lbs. was chosen as the pilot's weight and for the most aft C.G. condition, 160 lbs. was used as the weight of the pilot. It is quite obvious that for an aircraft of such low empty weight, the C.G. is affected considerably by the weight of the pilot.

The horizontal datum (from which the X distances are measured) is the nose of the aircraft (Station 0). The vertical datum (from which the Z distances are measured) is the ground line. The lateral datum (from which the Y distances are measured) is the Centerline of the aircraft.

GROUP WEIGHT STATEMENT

ESTIMATED - CALCULATED - ACTUAL

(Cross out those not applicable)

CONTRACT NO. NO nr 2360(00)

AIRPLANE, GOVERNMENT NO. YAO-3(G)

AIRPLANE, CONTRACTOR NO. GA-468

MANUFACTURED BY GOODYEAR AIRCRAFT CORPORATION

		MAIN	AUXILIARY
ENGINE	MANUFACTURED BY	Nelson	
	MODEL	H-63A Modified	
	NO.		
PROPELLER	MANUFACTURED BY	U.S. Propellers	
	DESIGN NO.	Model 380-31 47 In. Dia., Wood	
	NO.		

AN-9103-D
NAME (illegible signature)
DATE December 17, 1960

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE 2.01.010
MODEL GA-466
REPORT GER-2061

No.	Item					
1	WING GROUP					46.9
2	CENTER SECTION - BASIC STRUCTURE					
3	INTERMEDIATE PANEL - BASIC STRUCTURE				23.5	
4	OUTER PANEL - BASIC STRUCTURE (INCL. TIPS ____ LBS.)					
5						
6	SECONDARY STRUCTURE (INCL. WINGFOLD MECHANISM ____ LBS.)				11.0	
7	AILERONS (INCL. BALANCE WEIGHT ____ LBS.)				5.3	
8	FLAPS - TRAILING EDGE				3.4	
9	- LEADING EDGE					
10	SLATS					
11	SPOILERS					
12	SPEED BRAKES					
13	INFLATION AIR				3.7	
14						
15	TAIL GROUP					18.6
16	STABILIZER - BASIC STRUCTURE				3.0	
17	FINS - BASIC STRUCTURE (INCL. DORSAL ____ LBS.)				3.3	
18	SECONDARY STRUCTURE (STAB. & FINS)				3.4	
19	ELEVATOR (INCL. BALANCE WEIGHT ____ LBS.)				2.5	
20	RUDDERS (INCL. BALANCE WEIGHT 4.5 LBS.)				6.0	
21	INFLATION AIR				.4	
22						
23	BODY GROUP					41.2
24	FUSELAGE OR HULL - BASIC STRUCTURE (ENVELOPE)				20.4	
25	BOOMS - BASIC STRUCTURE					
26	SECONDARY STRUCTURE - FUSELAGE OR HULL (COCKPIT & FURNISHINGS)				18.9	
27	- BOOMS					
28	- SPEEDBRAKES					
29	- DOORS, PANELS & MISC.					
30	INFLATION AIR				1.9	
31	ALIGHTING GEAR GROUP - LAND (TYPE: UNICYCLE)					23.0
32–33	LOCATION	WHEELS, BRAKES TIRES, TUBES, AIR	STRUCTURE	CONTROLS		
34	MAIN	5.3	16.1		21.4	
35	TAIL SKID				.7	
36	WING TIP SKIDS				.9	
37						
38						
39						
40	ALIGHTING GEAR GROUP - WATER					
41	LOCATION	FLOATS	STRUTS	CONTROLS		
42						
43						
44						
45						
46	SURFACE CONTROLS GROUP					2.7
47	COCKPIT CONTROLS - STICK				.7	
48	AUTOMATIC PILOT					
49	SYSTEM CONTROLS (INCL. POWER & FEEL CONTROLS ____ LBS.)				2.0	
50						
51	ENGINE SECTION OR NACELLE GROUP					9.5
52	INBOARD					
53	CENTER					
54	OUTBOARD					
55	DOORS, PANELS & MISC.					
56						
57	TOTAL (TO BE BROUGHT FORWARD)					141.9

AN-9103-D
NAME ____________
DATE December 19, 1970

GROUP WEIGHT STATEMENT
WEIGHT EMPTY

PAGE 2.01.010
MODEL QA-160
REPORT 9001

No.	Item	Auxiliary	Main		Total
1	PROPULSION GROUP				68.5
2		AUXILIARY	MAIN		
3	ENGINE INSTALLATION			56.2	
4	AFTERBURNERS (IF FURN. SEPARATELY)				
5	ACCESSORY GEAR BOXES & DRIVES				
6	SUPERCHARGERS (FOR TURBO TYPES)				
7	AIR INDUCTION SYSTEM				
8	EXHAUST SYSTEM				
9	COOLING SYSTEM				
10	LUBRICATING SYSTEM				
11	TANKS				
12	COOLING INSTALLATION				
13	DUCTS, PLUMBING, ETC.				
14	FUEL SYSTEM			5.6	
15	TANKS - PROTECTED		3.1		
16	- UNPROTECTED				
17	PLUMBING, ETC.		2.5		
18	WATER INJECTION SYSTEM				
19	ENGINE CONTROLS			.8	
20	STARTING SYSTEM				
21	PROPELLER INSTALLATION			5.9	
22					
23					
24	AUXILIARY POWER PLANT GROUP				
25	INSTRUMENTS & NAVIGATIONAL EQUIPMENT GROUP				7.6
26	~~HYDRAULIC~~ & PNEUMATIC GROUP				11.9
27					
28					
29	ELECTRICAL GROUP				2.0
30	BATTERY			1.4	
31	COCKPIT LIGHT			.6	
32	ELECTRONICS GROUP				
33	EQUIPMENT				
34	INSTALLATION				
35					
36	ARMAMENT GROUP (INCL. GUNFIRE PROTECTION LBS.)				
37	FURNISHINGS & EQUIPMENT GROUP				
38	ACCOMMODATIONS FOR PERSONNEL				
39	MISCELLANEOUS EQUIPMENT				
40	FURNISHINGS				
41	EMERGENCY EQUIPMENT				
42					
43	AIR CONDITIONING & ANTI-ICING EQUIPMENT GROUP				
44	AIR CONDITIONING				
45	ANTI-ICING				
46					
47	PHOTOGRAPHIC GROUP				
48	AUXILIARY GEAR GROUP				
49	HANDLING GEAR				
50	ARRESTING GEAR				
51	CATAPULTING GEAR				
52	ATO GEAR				
53					
54					
55	MANUFACTURING VARIATION				
56	TOTAL FROM PG. 2				141.9
57	WEIGHT EMPTY				231.9

AN-9103-D
NAME ____
DATE December 19, 1970

GROUP WEIGHT STATEMENT
USEFUL LOAD & GROSS WEIGHT

PAGE 2.01.030
MODEL QA-460
REPORT QAM-2061

1	LOAD CONDITION						
2							
3	CREW (NO. 1)						200.0
4	PASSENGERS (NO.)						
** 5	FUEL & OIL	Type **	Gals.				
6	UNUSABLE						
7	INTERNAL	at 6.09#/Gal.	19.4				118.1
8							
9							
10	EXTERNAL						
11							
12	BOMB BAY						
13							
** 14	OIL						**
15	TRAPPED						
16	ENGINE						
17							
18	FUEL TANKS (LOCATION)						
19	WATER INJECTION FLUID (GALS)						
20							
21	BAGGAGE						
22	CARGO						
23							
24	ARMAMENT						
25	GUNS (Location)	Fix. or Flex.	Qty.	Cal.			
26							
27							
28							
29							
30							
31							
32	AMMUNITION						
33							
34							
35							
36							
37							
38							
39	INSTALLATIONS (BOMB, TORPEDO, ROCKET, ETC.)						
*40	BOMB OR TORPEDO RACKS						
41							
42							
43							
44							
45							
46	EQUIPMENT						
47	PYROTECHNICS						
48	PHOTOGRAPHIC						
49							
*50	OXYGEN						
51							
52	MISCELLANEOUS						
53							
54							
55	USEFUL LOAD						318.1
56	WEIGHT EMPTY						231.9
57	GROSS WEIGHT						550.0

*If not specified as weight empty.

** Fuel & oil are pre-mixed (by volume) in ratio 16 fuel/1 oil.

AN-9103-D

NAME ______

DATE December 17, 1960

GROUP WEIGHT STATEMENT
DIMENSIONAL & STRUCTURAL DATA

PAGE 2.01.070

MODEL OA-460

REPORT OZR-7001

		Main Floats	Aux. Floats	Booms	Fuse. or Hull	Nacelles Inboard	Nacelles Center	Nacelles Outboard
1	LENGTH - OVERALL (FT.) 21.09	HEIGHT - OVERALL - STATIC (FT.)						8.1
2								
3	LENGTH - MAX. (FT.)				20.17			
4	DEPTH - MAX. (FT.)				2.90			
5	WIDTH - MAX. (FT.)				2.23			
6	WETTED AREA (SQ. FT.)							
*7	FLOAT OR HULL DISPL. - MAX (LBS.)							
8	FUSELAGE VOLUME (CU. FT.)		PRESSURIZED			TOTAL		

		Wing	H. Tail	V. Tail
9				
10	GROSS AREA (SQ. FT.)	110.00	20.0	15.31
11	WEIGHT/GROSS AREA (LBS./SQ. FT.)	5.00		
12	SPAN (FT.)	22.0	6.67	
13	FOLDED SPAN (FT.)			
14				
15	SWEEPBACK - AT 25% CHORD LINE (DEGREES)	0		
16	- AT % CHORD LINE (DEGREES)	0		
**17	THEORETICAL ROOT CHORD - LENGTH (INCHES)	60		
18	- MAX. THICKNESS (INCHES)	9		
***19	CHORD AT PLANFORM BREAK - LENGTH (INCHES)	-		
20	- MAX. THICKNESS (INCHES)	-		
***21	THEORETICAL TIP CHORD - LENGTH (INCHES)	60		
22	- MAX. THICKNESS (INCHES)	9		

23 DORSAL AREA, INCLUDED IN (FUSE.) (HULL) (V. TAIL) AREA (SQ. FT.)

24 TAIL LENGTH - 25% MAC WING TO 25% MAC H. TAIL (FT.) 14.23

25	AREAS (SQ. FT.) Flaps	L.E.	T.E. 10.00		
26	Lateral Controls	Slots	Spoilers	Ailerons 12.00	
27	Speed Brakes	Wing	Fuse. or Hull		
28					
29					

30 ALIGHTING GEAR UNICYCLE (LOCATION)

31 LENGTH - OLEO EXTENDED - ℄ AXLE TO ℄ TRUNNION (INCHES)

32 OLEO TRAVEL - FULL EXTENDED TO FULL COLLAPSED (INCHES)

33 FLOAT OR SKI STRUT LENGTH (INCHES)

34 ARRESTING HOOK LENGTH - ℄ HOOK TRUNNION TO ℄ HOOK POINT (INCHES)

35 HYDRAULIC SYSTEM CAPACITY (GALS.)

		Location	No. Tanks	****Gals. Protected	No. Tanks	****Gals. Unprotected
36	FUEL & LUBE SYSTEMS					
37	Fuel - Internal	Wing				
38		Fuse. or Hull			1	20
39	- External					
40	- Bomb Bay					
41						
42	Oil					
43						
44						

		Fuel in Wings (Lbs.)	Stress Gross Weight	Ult. L.F.
45	STRUCTURAL DATA - CONDITION			
46	FLIGHT	0	550	6.375
47	LANDING			
48				
49	MAX. GROSS WEIGHT WITH ZERO WING FUEL			
50	CATAPULTING			
51	MIN. FLYING WEIGHT			
52	LIMIT AIRPLANE LANDING SINKING SPEED (FT./SEC.)			

53 WING LIFT ASSUMED FOR LANDING DESIGN CONDITION (%W)

54 STALL SPEED - LANDING CONFIGURATION - POWER OFF (KNOTS)

55 PRESSURIZED CABIN - ULT. DESIGN PRESSURE DIFFERENTIAL - FLIGHT (P.S.I.)

56

57 AIRFRAME WEIGHT (AS DEFINED IN AN-W-11) (LBS.)

*Lbs. of sea water @ 64 lbs./cu. ft.
**Parallel to ℄ at ℄ airplane.

***Parallel to ℄ airplane.
****Total usable capacity.

PREPARED ________
CHECKED ________
DATE ____ December 19
REV DATE ________

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TABLE IV

MOMENT OF INERTIA CALCULATIONS

	LBS.	IN.			IN.-LBS.		
ITEM	W	X	Y	Z	WX	WY	WZ
Wing Group	(46.9)	(87.8)	0	(54.3)	(4120)	0	(2549)
Envelope	23.5	88.5	0	56.2	2053	0	1433
Ailerons (2)	3.5	108.0	±96.0	50.7	381	0	177
Flap	2.4	108.8	0	50.7	261	0	122
Brace Wire	2.5	94.4	0	50.0	236	0	125
Hinge	1.5	105.0	±95.0	51.8	158	0	78
Main Brace Patch	3.5	82.0	±60.0	54.0	287	0	189
Strap & Patches	2.3	105.0	±50.0	50.0	242	0	115
Bungee Cord	.5	97.0	±96.0	56.3	49	0	28
Paint	1.5	82.0	0	56.2	123	0	84
Inflation Air 7 psi	3.7	89.2	0	53.5	330	0	198
Tail Group	18.6	(233.1)	(0)	(66.9)	(4335)	(0)	(1245)
Stabilizer	3.0	228.6	0	51.7	686	0	155
Elevator	2.5	245.5	0	51.7	614	0	129
Hinge, Control Horn, Patches	1.7	239.4	0	51.7	407	0	88
Fin	3.3	222.1	0	70.7	733	0	233
Rudder Structure	1.5	248.0	0	76.2	372	0	114
Rudder Balance Weight	4.5	233.8	±6.0	86.5	1052	0	389
Hinge, Control Horn, Patches	1.0	239.4	0	72.0	239	0	72
Brace Wires	.3	160.0	0	57.0	48	0	17
Paint	.4	229.5	0	60.0	92	0	24
Inflation Air	.4	229.5	0	60.0	92	0	24
Body Group	(41.2)	(90.8)	0	(34.2)	(3741)	0	(1409)
Envelope	16.0	130.1	0	39.2	2082	0	627
Paint & Inflation	5.8	130.1	0	39.2	755	0	227
Cockpit (Incl. Air, Paint, Patches)	19.4	46.6	0	28.6	904	0	555
Alighting Gear Group	(23.0)	(72.6)		(19.7)	(1669)	0	(454)
Main Gear	15.3	63.5	0	17.7	972	0	271
Tail Skid	.7	231.3	0	34.0	162	0	24
Wing Tip Skids	.9	98.0	±125.0	42.0	88	0	38
Reinforcement	6.1	73.3	0	19.8	447	0	121
Surface Controls Group	2.7	(58.9)	0	(34.1)	(159)	0	(92)
Control Stick	.7	24.0	0	32.0	17	0	22
Cables	2.0	71.0	0	35.0	142	0	70

2

TABLE IV

ENT OF INERTIA CALCULATIONS

	IN.-LBS.			LB.-IN.²/1000					
Z	WX	WY	WZ	WX^2	WY^2	WZ^2	I_x	I_y	I_z
(34.3)	(4120)	0	(2549)	(367.3)	(69.1)	(130.9)	(202.7)	(9.8)	(210.2)
56.2	2053	0	1433	165.3	0	80.5	152.0	5.8	167.4
50.7	301	0	177	41.5	32.3	9.9	1.5	.1	1.6
50.7	261	0	122	28.4	0	6.2	2.8		2.8
50.0	236	0	125	22.3	0	6.3	3.0	.5	3.0
51.8	158	0	78	16.6	13.8	4.0	--	.7	.7
54.0	287	0	189	23.5	12.6	10.2	.7	.7	.7
50.0	242	0	115	23.4	5.8	5.8	.4	.4	.4
56.3	49	0	28	4.8	4.6	1.6	.1	.1	.1
56.2	123	0	84	10.1	0	4.7	9.5	.4	9.8
55.5	330	0	198	29.4	0	10.6	22.7	1.1	23.7
(66.9)	(4335)	(0)	(1245)	(1013.0)	(.2)	(87.0)	(4.9)	(3.1)	(6.1)
51.7	606	0	155	156.8	0	8.0	1.6	.4	1.8
51.7	614	0	129	150.7	0	6.7	1.4	.3	1.5
51.7	407	0	88	97.4	0	4.5	.9	.1	1.1
70.7	733	0	233	162.8	0	16.5	.5	1.2	.4
76.2	372	0	114	92.3	0	8.7	.2	.5	.7
86.5	1052	0	309	245.9	.2	33.6	.1	.1	.1
72.0	239	0	72	57.2	0	5.2	.1	.4	.3
57.0	48	0	17	7.7	0	1.0			
60.0	92	0	24	21.1	0	1.4	.1	.1	.2
60.0	92	0	24	21.1	0	1.4			
(34.2)	(3741)	0	(1409)	(411.2)	0	(49.4)	(6.5)	(71.1)	(69.7)
39.2	2082	0	627	270.9	0	24.6	2.6	55.5	55.5
39.2	755	0	227	98.2	0	8.9			
28.6	904	0	555	42.1	0	15.9	3.9	15.6	14.2
(19.7)	(1669)	0	(454)	(140.5)	(14.1)	(9.6)	(2.5)	(2.4)	(2.5)
17.7	972	0	271	61.7	0	4.8	2.0	2.0	2.0
34.0	162	0	24	37.5	0	.8	-	-	-
42.0	88	0	38	8.6	14.1	1.6	-	-	-
19.8	447	0	121	32.7	0	2.4	.5	.4	.5
(34.1)	(159)	0	(92)	(10.5)	0	(3.2)	(2.7)	(8.8)	(8.8)
32.0	17	0	22	.4	0	.7	-	-	-
35.0	142	0	70	10.1	0	2.5	2.7	8.8	8.8

1

TABLE IV (Continued)

MOMENT OF INERTIA CALCULATIONS

	LBS.	IN.			IN.-LBS.	
ITEM	W	X	Y	Z	WX	WY
Engine Section	(9.5)	(107.3)	0	(64.0)	(1019)	0
Engine Mount	8.8	107.8	0	64.7	949	0
Fuselage Patch	.7	100.0	0	56.0	70	0
Propulsion Group	(68.3)	(107.3)	0	(79.0)	(7352)	0
Magneto	6.9	119.5	0	81.5	825	0
Engine	45.8	108.8	0	82.2	4983	0
Carburetor	2.2	107.0	0	87.5	235	0
Spark Plugs	1.3	107.5	0	82.3	140	0
Propeller	4.1	99.4	0	82.7	408	0
Hub, Flange, Bolts	1.8	99.4	0	82.7	179	0
Fuel Cell	3.1	74.5	0	30.2	231	0
Hoses	1.5	103.5	0	58.0	155	0
Shut-off Valve, Press. Reg., Fuel Coupling & Fttg.	1.0	106.0	0	63.0	106	0
Engine Controls	.8	112.5	0	65.5	90	0
Instruments	(7.6)	(23.8)	(-1.8)	(43.0)	(181)	(-14)
Air Pressure Gage	.4	19.0	-3.5	43.0	8	-1
Airspeed Indicator	.7	19.0	+7.5	42.0	13	+5
Pitot Tube	.3	0	+13.7	34.0	0	+4
Compass	.5	19.0	+3.0	42.5	10	+2
Tachometer	1.9	19.0	-8.8	42.0	36	-17
Altimeter	1.5	19.0	+5.0	42.0	29	+8
Cylinder Head Temp.	1.5	19.0	-6.0	42.0	29	-9
Thermocouple	.8	70.0	-8.0	55.0	56	-6
Pneumatic Group	(11.9)	(117.6)	0	(80.9)	(1400)	0
Compressor	5.4	122.6	0	85.8	662	0
Compressor Valve	4.9	119.0	0	85.8	583	0
Relief Valve	.5	107.6	0	50.5	54	0
Check Valve	.3	107.6	0	50.5	32	0
Hose & Fitting	.5	85.0	0	50.0	43	0
Control Cables	.3	85.0	0	50.0	26	0
Electrical Group	(2.0)	(32.0)	(+6.5)	(28.5)	(64)	(+13)
Battery	1.4	30.0	+5.0	25.0	42	+7
Cockpit Light	.6	37.0	+10.0	36.0	22	+6
Total Weight Empty	231.9	103.67	.004	56.56	24040	-1

2

TABLE IV (Continued)

OMENT OF INERTIA CALCULATIONS

	IN.-LBS.			LB.-IN.²/1000					
Z	WX	WY	WZ	WX^2	WY^2	WZ^2	I_x	I_y	I_z
(64.0)	(1019)	0	(608)	(109.3)	0	(39.0)	(1.0)	(2.6)	(1.6)
64.7	949	0	569	102.3	0	36.8	1.0	2.6	1.6
56.0	70	0	39	7.0	0	2.2			
(79.0)	(7352)	0	(5411)	(793.9)	0	(436.5)	(3.9)	(3.4)	(5.2)
81.5	825	0	562	98.6	0	45.8			
82.2	4993	0	3765	542.2	0	307.5	1.7	2.1	3.6
87.5	235	0	193	25.1	0	16.9			
82.3	140	0	107	15.1	0	8.8			
82.7	408	0	339	40.6	0	28.0	1.8	.9	.9
82.7	179	0	149	17.8	0	12.3			
30.2	231	0	94	17.2	0	2.8			
58.0	155	0	87	16.0	0	5.0	.4	.4	.7
63.0	106	0	63	11.2	0	4.0			
65.5	90	0	52	10.1	0	3.4	-	-	-
(43.0)	(181)	(-14)	(327)	(6.4)	(0.3)	(14.1)	(..1)	(.1)	(.1)
43.0	8	-1	17	.2	-	.7			
42.0	13	+5	29	.2	-	1.2			
34.0	0	+4	10	0	.1	.3			
42.5	10	+2	21	.2	-	.9	.1	.1	.1
42.0	36	-17	80	.7	.1	3.4			
42.0	29	+8	63	.6	-	2.6			
42.0	29	-9	63	.6	.1	2.6			
55.0	56	-6	44	3.9	-	2.4			
(80.9)	(1400)	0	(963)	(165.7)	0	(79.9)	(1.2)	(1.7)	(1.7)
85.8	662	0	463	81.2	0	39.7	.4	.6	.6
85.8	583	0	420	69.4	0	36.0	.4	.5	.5
50.5	54	0	25	5.8	0	1.3	.1	.2	.2
50.5	32	0	15	3.4	0	.8	.1	.1	.1
50.0	43	0	25	3.7	0	1.3	.1	.2	.2
50.0	26	0	15	2.2	0	.8	.1	.1	.1
(28.5)	(64)	(+13)	(57)	(2.1)	(.1)	(1.7)	-	-	-
25.0	42	+7	35	1.3	-	.9	-	-	-
36.0	22	+6	22	.8	.1	.8	-	-	-
56.56	24040	-1	13115	3019.9	83.8	859.3	225.5	103.0	305.9

PREPARED
CHECKED
DATE December 19, 196
REV DATE

1

TABLE IV (Continued)

MOMENT OF INERTIA CALCULATIONS

	LBS.	IN.			IN.-LBS.			
ITEM	W	X	Y	Z	WX	WY	WZ	W
					MOST AFT C.G.			
Weight Empty	231.9	103.67	-.004	56.33	24040	-1	13115	3
Pilot	160.0	40.0	0	37.2	6400	0	5952	
Fuel	12.4	74.6	0	25.0	925	0	310	
Gross Weight	404.3	77.58	-.0025	47.93	31365	-1	19377	3 -2

Most Aft C.G. Location in Percent MAC = $\left(\frac{77.6 - 57.7}{60.0}\right) 100 = 33.2$

	LBS.	IN.			IN.-LBS.			
ITEM	W	X	Y	Z	WX	WY	WZ	W
					MOST FWD C.G.			
Weight Empty	231.9	103.67	-.004	56.55	24040	-1	13115	3
Pilot	240.0	40.00	0	37.20	9600	0	8928	
Fuel	78.1	74.60	0	29.20	5826	0	2202	
Gross Weight	550.0	71.76	-.002	44.08	39466	-1	24245	-2

Most FWD C.G. Location in Percent MAC = $\left(\frac{71.8 - 57.7}{60.0}\right) 100 = 23.5$

Length of MAC = 60.0 inches.
L.E. of MAC is located at Station 57.7.

2

TABLE IV (Continued)

MENT OF INERTIA CALCULATIONS

	IN.-LBS.			LB.-IN.^{2}/1000					
Z	WX	WY	WZ	WX^2	WY^2	WZ^2	I_x	I_y	I_z
	MOST AFT C.G.								
56.33	24040	-1	13115	3019.9	83.9	859.3	225.5	103.0	305.9
37.2	6400	0	5952	256.0	--	221.4	16.9	23.2	14.2
25.0	925	0	310	69.0	--	7.8	1.7	.4	2.0
47.93	31365	-1	19377	3344.9	83.8	1088.5	244.1	128.6	322.1
				-2433.3	-0	-928.7	83.8	911.6	911.6
				911.6	83.9	159.8	159.8	159.9	83.8
00 = 33.2				Lb. In.^{2}/1000			487.7	1200.0	1317.5
				Slug Ft.^{2}			105.3	259.0	294.4
	MOST FWD C.G.								
56.33	24040	-1	13115	3019.9	83.8	859.3	225.5	103.0	305.9
37.20	9600	0	8928	384.0	0	332.1	30.7	42.9	24.4
23.20	5826	0	2202	434.6	0	62.1	10.9	3.1	3.1
44.08	39466	-1	24245	3838.5	83.8	1253.5	267.1	149.0	333.4
				-2832.1	0	-1068.7	83.8	1006.4	1006.4
				1006.4	83.8	184.8	184.8	184.8	83.8
				Lb. In.^{2}/1000			535.7	1340.2	1423.6
				Slug Ft.^{2}			115.6	289.3	307.3
00 = 23.5									

WING LOADS

The wing is of Airmat construction and is a single-piece structural unit forward of the aileron hinge line. External support for the wing is provided by brace cables connected between the upper wing surface and engine mount pylon. The lower wing surface brace cables are connected to the landing gear and fuselage. The wing is attached to the fuselage by fabric straps at the rear cockpit bulkhead and through the engine mount attachment cables at the trailing edge of the wing.

The sign convention used for the wing load calculations is shown on page 2.02.020.

The shears, moments, and torques for the critical wing symmetrical conditions A_2 and C_2 are found on page 2.02.170 through 2.02.260. Unsymmetrical conditions were investigated and found not to be critical.

No torque results from the chordwise force.

PREPARED BY: J.R.J.
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GOODYEAR AIRCRAFT

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WING LOADS

Sign Convention

Loads Positive As Shown
Moments R.H. Rule

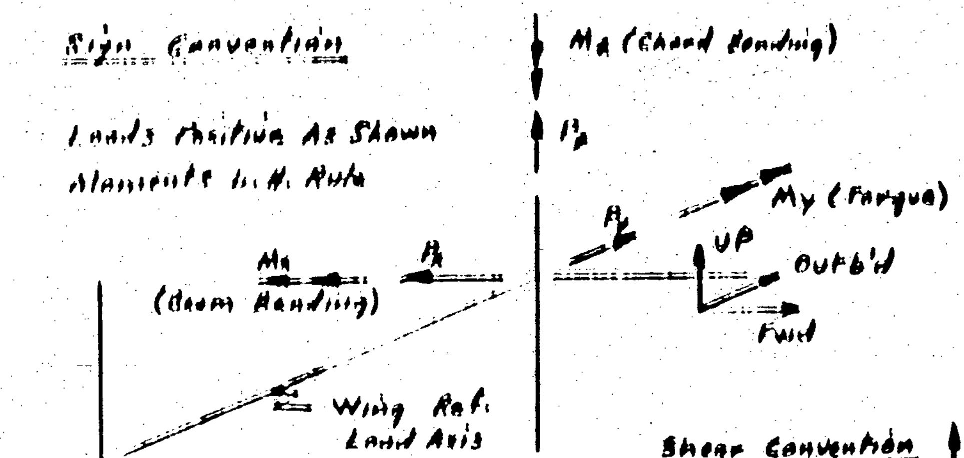


Figure 2a

Shear Convention

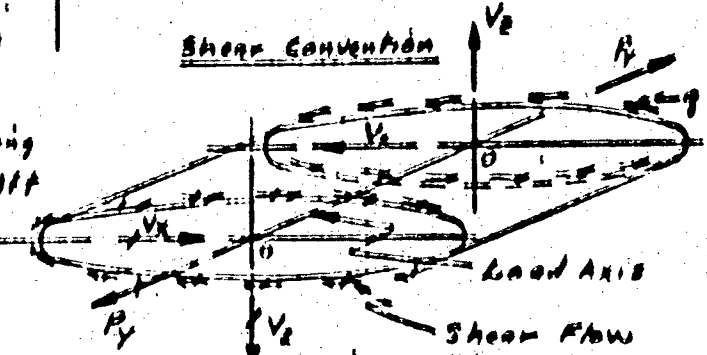
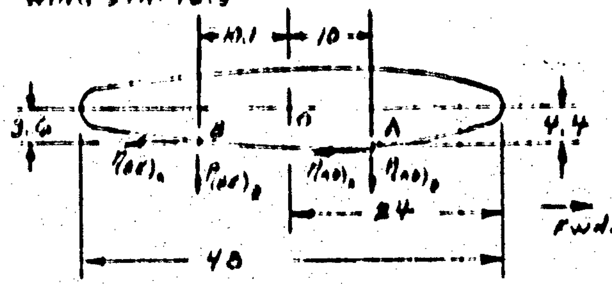


Figure 2b

P_y = Axial Tension
V_x = Chord Shear
V_z = Beam Shear

Outb'd Brace Wire Attachments
WING STA. 16.3



Note:
Point O = Assumed Load Axis
Tension In Cables Positive

Figure 2c

Inb'd Brace Wire Attachment
WING STA. 43.6

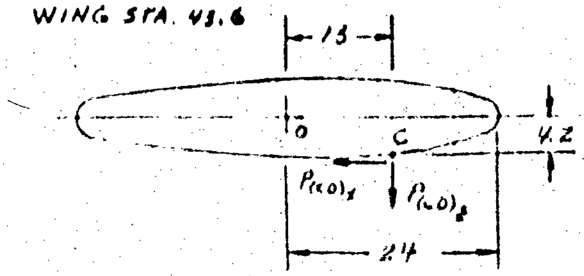


Figure 2d

218-83 (3)

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WING LOADS

Calculation Of Shears, Moments And Torques From The Airloads

Using The Spanwise Lift Distribution Curve (C_{l_c} ~ Semi-Span) And The Spanwise Drag Distribution Curve (C_d/b_2 ~ Semi-Span) of Reference (4) The Normal Force Coefficient (C_N) And The Chord Force Coefficient (C_c) Are Calculated In Table V. Using The Equations Listed Below:

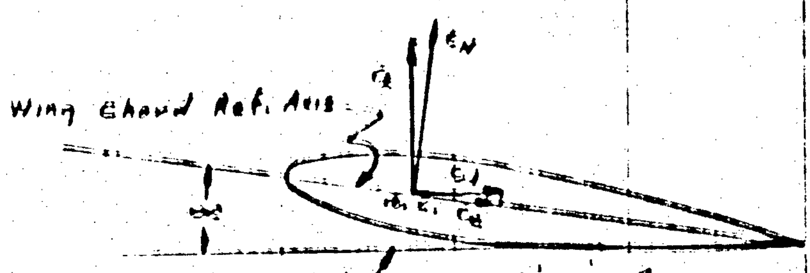

Figure 3

$$C_N = C_l \cos\alpha + C_d \sin\alpha$$
$$C_c = C_d \cos\alpha - C_l \sin\alpha$$

Where: C_l = Section Lift Coefficient
C_d = Section Drag Coefficient

The General Equation For Airfoil Lift Coefficient Is

$$C_L = \frac{L}{qS}$$

Where: L = Total Lift of The Airfoil - Lbs.
S = Airfoil Area - Ft^2
q = $\rho/2\,V^2$ Dynamic Pressure - Lbs/Ft^2
ρ = .002378 Slugs / Ft^3
V = Velocity In Ft/sec.

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GOODYEAR AIRCRAFT

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WING LOADS

Calculation of Shears, Moments, And Torques From The Airloads

The Section Lift Coefficient over the Elementary Area

or $\dfrac{dL}{q\,dS}$

Where $dS = c\,dy = c\,\Delta b$
c = Airfoil Chord
Δb = An Elementary Spanwise Distance

Then
$dL = C_\ell\, c\, q\, \Delta b$

Similarly
$dD = C_d\, c\, q\, \Delta b$

Resolving These Into Normal And Chordwise Force Distribution Curves, dN And dC, And Using A Semi-graphical Method Of Intergration The Shears And Moments were Calculated. These Calculations Are Shown In Tables VI, VII, VIII, & IX.

The Torques About The Reference Axis Was Computed By Transferring Normal Force From The Quarter-Chord To The Reference Axis. The Calculation For The Torques Appears In Tables VI & VII.

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Calculation Of Shears, Moments, And Torques From The Airloads

Table V

Col.	①	②	③	④	⑤	⑥	⑦	⑧	⑨
Item	Sta.	α Degrees	Cos α	Sin α	C_{l_t}	C_l	C_{DT}/C_{LT}	C_d	$C_l \cos\alpha$
Ref.						⑤/5		⑦×11	③×⑥
Condition - A_z HAA Symmetrical Maneuver	132	13.8	.97113	.23853	0	=	.0015	.0165	0
	120	↑	↑	↑	3.70	0.740	.0056	.0616	0.717
	108				5.05	1.010	.0068	.0748	0.981
	96				5.90	1.180	.0112	.1232	1.146
	84				6.50	1.300	.0133	.1463	1.262
	76.3				6.79	1.358	.0144	.1584	1.319
	72				6.95	1.390	.0150	.1650	1.350
	60				7.25	1.450	.0163	.1793	1.408
	48				7.49	1.498	.0172	.1892	1.455
	43.6				7.54	1.508	.0175	.1925	1.464
	36				7.64	1.528	.0179	.1969	1.484
	24				7.76	1.552	.0184	.2024	1.507
	12	↓	↓	↓	7.83	1.566	.0186	.2046	1.521
	0	13.8	.97113	.23853	7.85	1.570	.0188	.2068	1.525

Condition - C_z LAA Symmetrical Maneuver	132	3.4	.99824	.05931	0	—	.0015	.0165	—
	120	↑	↑	↑	1.95	0.390	.0025	.0275	0.389
	108				2.62	0.524	.0034	.0374	0.523
	96				3.05	0.610	.0041	.0451	0.609
	84				3.35	0.670	.0047	.0517	0.669
	76.3				3.50	0.700	.0050	.0550	0.699
	72				3.58	0.716	.0052	.0572	0.715
	60				3.72	0.744	.0055	.0605	0.743
	48				3.85	0.770	.00575	.0633	0.769
	43.6				3.90	0.780	.00585	.0644	0.779
	36				3.93	0.786	.0059	.0649	0.785
	24				3.98	0.796	.0060	.0660	0.795
	12	↓	↓	↓	3.99	0.798	.0060	.0660	0.797
	0	3.4	.99824	.05931	4.00	0.800	.0060	.0660	0.799

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WING LOADS

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Table V

⑥	⑦	⑧	⑨	⑩	⑪	⑫	⑬	⑭
C_A	$C_{DT}/_{\ell_{1/2}}$	C_d	$C_\ell \cos\alpha$	$C_d \sin\alpha$	C_N	$C_\ell \sin\alpha$	$C_d \cos\alpha$	C_c
⑤/S		⑦ X 11	③ x ⑥	④ x ⑧	⑨ + ⑩	④ x ⑥	③ x ⑧	⑬ - ⑫
—	.0015	.0165	0	.0039	0.0039	—	.0160	+.0160
740	.0056	.0616	0.717	.0147	0.7317	.1105	.0598	-.1107
010	.0088	.0968	0.981	.0231	1.0041	.2499	.0940	-.1469
180	.0112	.1232	1.146	.0294	1.1754	.2815	.1195	-.1817
300	.0133	.1463	1.262	.0349	1.2969	.3131	.1421	-.1639
358	.0144	.1584	1.319	.0378	1.3563	.3237	.1538	-.1701
310	.0150	.1650	1.350	.0394	1.3374	.3315	.1602	-.1714
459	.0163	.1793	1.408	.0428	1.4508	.3457	.1741	-.1719
493	.0172	.1892	1.455	.0451	1.5001	.3573	.1837	-.1736
503	.0175	.1925	1.464	.0459	1.5099	.3597	.1869	-.1728
528	.0179	.1969	1.484	.0470	1.5310	.3645	.1912	-.1733
552	.0184	.2024	1.507	.0483	1.5553	.3702	.1956	-.1736
566	.0186	.2046	1.521	.0488	1.5698	.3715	.1987	-.1749
570	.0188	.2068	1.525	.0493	1.5743	.3745	.2008	-.1737

—	.0015	.0165	—	.0010	.0010	—	.0165	.0165
390	.0025	.0275	0.389	.0016	.3906	.0231	.0275	.0044
524	.0034	.0374	0.523	.0022	.5252	.0311	.0373	.0062
610	.0041	.0451	0.609	.0027	.6117	.0362	.0450	.0088
670	.0047	.0517	0.669	.0031	.6721	.0397	.0516	.0119
700	.0050	.0550	0.699	.0033	.7023	.0415	.0549	.0134
.716	.0052	.0572	0.715	.0034	.7184	.0425	.0571	.0146
.744	.0055	.0605	0.743	.0036	.7466	.0441	.0604	.0163
.770	.00575	.0633	0.769	.0038	.7728	.0457	.0632	.0175
.780	.00585	.0644	0.779	.0038	.7828	.0463	.0643	.0180
.786	.0059	.0649	0.785	.0038	.7890	.0466	.0648	.0182
.796	.0060	.0660	0.795	.0039	.7989	.0472	.0659	.0187
.798	.0060	.0660	0.797	.0039	.8009	.0473	.0659	.0183
.800	.0060	.0660	0.799	.0039	.8029	.0474	.0659	.0185

1

Calculation of Shears, Moments, and Torques For Normal
From the Airloads

Table VI

Condition - A2 HAA Symmetrical Maneuver q = 9.65#/Ft²

Col.	①	②	③	④	⑤	⑥	⑦	⑧	
Item	Sta	C_N	C_N*C	Sum	$\frac{\Delta b}{Ft}$	$\Delta(C_N C \Delta b)$	$C_N C \Delta b$	Shear	S
Ref.	In		②·5	③+③		½④⑤	Σ ⑥	9.65⑦	⑧
	192	≅ 0	0				0	0	
				3.669	1.0	1.835			1
	180	.7337	3.669				1.835	17.71	
				8.690	1.0	4.345			4
	168	1.0041	5.021				6.180	59.64	
				10.898	1.0	5.449			17
	96	1.1754	5.877				11.629	112.22	
				12.362	1.0	6.181			2
	84	1.2909	6.455				17.810	171.87	
				13.239	.64	4.236			3
	76.3	1.3568	6.784				22.056	212.84	
				13.731	.36	2.472			4
	72	1.3894	6.947				24.528	236.70	
				14.201	1.0	7.101			5
	60	1.4508	7.254				31.629	305.22	
				14.755	1.0	7.378			6
	48	1.5001	7.501				39.007	376.42	
				15.051	.36	2.709			77
	43.6	1.5099	7.550				41.716	402.56	
				15.205	.64	4.866			8
	36	1.5310	7.655				46.582	449.52	
				15.432	1.0	7.716			97
	24	1.5553	7.777				54.298	523.98	
				15.626	1.0	7.813			11
	12	1.5698	7.849				62.111	599.37	
				15.721	1.0	7.861			11
	0	1.5743	7.872				69.972	675.23	

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GOODYEAR AIRCRAFT

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and Torques For Normal Forces — Wing Loads

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Table VI

(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)
Δb Ft.	Δ(CnCab)	CnCab	Shear	Sum	ΔM	Moment Ft-Lb	Moment In-Lb	Torque In-Lb
	½(4)(5)	Σ(6)	9.65(7)	(8)+(8)n+1	½(5)(9)	Σ(10)	12×(11)	12×½×(8)
		0	0			0	0	0
1.0	1.835			17.71	8.86			
		1.835	17.71			8.86	106.3	155.4
1.0	4.345			77.35	38.68			
		6.180	59.64			47.54	570.5	536.8
1.0	5.449			171.86	85.93			
		11.629	112.22			133.47	1601.6	1010.0
1.0	6.181			284.09	142.05			
		17.810	171.87			275.52	3306.2	1546.8
.64	4.246			384.71	123.11			
		22.056	212.84			398.63	4783.6	1915.6
.36	2.472			449.54	80.92			
		24.528	236.70			479.55	5754.6	2130.3
1.0	7.101			541.92	270.96			
		31.629	305.22			750.51	9006.1	2747.0
1.0	7.378			681.64	340.82			
		39.007	376.42			1091.33	13096.1	3387.8
.36	2.709			778.98	140.22			
		41.716	402.56			1231.55	14778.6	3623.0
.64	4.866			852.08	272.67			
		46.582	449.52			1504.22	18050.6	4045.7
1.0	7.716			973.50	486.75			
		54.298	523.98			1990.97	23891.6	4715.8
1.0	7.813			1123.35	561.68			
		62.111	599.37			2552.65	30631.8	5394.3
1.0	7.861			1174.60	637.30			
		69.972	675.23			3189.95	38279.4	6077.1

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Calculations of Shears and Moments For Drag Forces From the Airloads

Table VII

Condition A. HAK Symmetrical Moment
q_0 = 9.65 #/ft²

Col.	(1)	(2)	(3)	(4)	(5)	(6)	(7)	(8)	(9)
Item	Sta.	C_f	$C_f C$	Sum	Δb Ft	$\Delta(C_f C \Delta b)$	$C_f C \Delta b$	Shear	
Ref.			(2)×5	$(3)_n + (3)_{n+1}$		½(5)(4)	Σ(6)	9.65(7)	(9)
	132	.2169	.0837				0	0	
				= .6835	1.3	= .2511			=
	120	= .1167	= .5835				= .2511	= 2.424	
				= 1.318	1.3	= .857			
	108	= .1464	= .7325				= .4107	= 4.118	
				= 1.544	1.3	= .772			=
	96	= .1619	= .8095				= 1.6827	= 16.238	
				= 1.6425	1.0	= .8212			=
	80	= .1683	= .8430				= 2.5039	= 24.117	
				= 1.6935	.64	= .5419			=
	76.3	= .1701	= .8505				= 3.0435	= 29.414	
				= 1.7375	.16	= .3374			
	72	= .1714	= .8570				= 3.3847	= 32.344	
				= 1.716	1.0	= .8580			=
	60	= .1718	= .8590				= 4.2139	= 40.664	
				= 1.727	1.0	= .8635			=
	48	= .1736	= .8680				= 5.0774	= 49.017	
				- 1.712	.36	- .3118			-
	[illegible]	- .1728	- .8640				- 5.3892	- 52.006	
				- 1.7305	.64	- .5533			-
	36	- .1733	- .8665				- 5.9430	- 57.350	
				- 1.7345	1.0	- .8672			-
	24	- .1736	- .8680				- 6.8102	- 65.718	
				- 1.742	1.0	- .8710			-
	12	- .1748	- .8740				- 7.6812	- 74.124	
				- 1.7425	1.0	- .8712			-
	0	- .1737	- .8685				- 8.5524	- 82.531	

GOODYEAR AIRCRAFT

Moments For Drag Forces. Wing Loads

Table VII

(6)	(7)	(8)	(9)	(10)	(11)	(12)
$\frac{h}{2}$(ΣΔh)	C_6 GΔb	Shear	Sum	Δm	Moment Ft-Lb	Moment In-Lb
$\frac{1}{2}$(5)(4)	Σ(6)		(3)+(7)	$\frac{1}{2}$(5)(9)	Σ(10)	12×(11)
	0	0			0	0
= .2511			= .0437	= 1.114		
	= .2511	= 2.413			= 1.214	= 14.57
= .656			= 11.217	= 5.608		
	= .9107	= 8.776			= 6.822	= 81.19
= .712			= 25.532	= 12.513		
	= 1.6327	= 15.936			= 19.335	= 232.36
= .8242			= 45.435	= 22.717		
	= 2.5375	= 24.117			= 41.552	= 498.66
= .5410			= 53.615	= 19.57		
	= 3.0485	= 29.413			= 56.717	= 643.63
= .3374			= 61.352	= 11.124		
	= 3.3557	= 32.330			= 67.443	= 819.11
= .8580			= 73.048	= 35.520		
	= 4.2137	= 40.614			= 104.367	= 1252.4
= .8635			= 89.661	= 44.823		
	= 5.0774	= 48.937			= 150.417	= 1805.0
= .3118			= 101.093	= 19.181		
	= 5.3892	= 52.056			= 172.377	= 2068.5
= .5533			= 109.256	= 34.994		
	= 5.9430	= 57.350			= 207.372	= 2488.5
= .8672			= 123.069	= 61.534		
	= 6.8102	= 65.718			= 268.906	= 3226.9
= .8710			= 139.842	= 69.921		
	= 7.6812	= 74.124			= 338.827	= 4065.9
= .8712			= 156.635	= 78.327		
	= 8.5524	= 82.531			= 417.154	= 5005.8

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Calculation of Shears, Moments, and Torques From the Airloads

Table VIII

Condition - C_2 LAA Symmetrical Maneuver
$q = 18.85$ #/ft²

Col.	①	②	③	④	⑤	⑥	⑦	
Item	Sta.	C_N	$C_N c$	Sum	Δb ft	$\Delta(C_N c \Delta b)$	$C_N c \Delta b$	
Ref	In		② x 5	③n + ③n+1		½ ④ ⑤	Σ ⑥	1
	132	≅ 0	0				0	
				1.953	1.0	.977		
	120	.3906	1.953				.977	
				4.579	1.0	2.289		
	108	.5252	2.626				3.267	
				5.685	1.0	2.843		
	96	.6117	3.059				6.110	1
				6.420	1.0	3.210		
	84	.6721	3.361				9.320	1
				6.873	.64	2.199		
	76.8	.7023	3.512				11.519	2
				7.104	.36	1.279		
	72	.7184	3.592				12.798	2
				7.325	1.0	3.663		
	60	.7466	3.733				16.461	3
				7.597	1.0	3.799		
	48	.7727	3.864				20.260	3
				7.778	.36	1.400		
	41.6	.7828	3.914				21.66	4
				7.854	.64	2.513		
	36.	.7880	3.940				24.173	4
				7.935	1.0	3.968		
	24	.7989	3.995				28.141	5
				8.000	1.0	4.000		
	12	.8009	4.005				32.141	6
				8.020	1.0	4.010		
	0	.8029	4.015				36.151	6

ments and Torques for Normal Forces — Wing Loads

Table VIII

(5)	(6)	(7)	(8)	(9)	(10)	(11)	(12)	(13)
Δb Ft.	$\Delta(C_N C b)$	$C_N C b$	Shear	Sum	ΔM	Moment Ft-Lb	Moment In-Lb	Torque In-Lb
	½(4)(5)	Σ(6)	18.85(7)	$(8)_n+(8)_{n-1}$	½(5)(9)	Σ(10)	12×(11)	12×¾(8)
		0	0			0	0	0
1.0	.977			18.43	9.22			
		.977	18.43			9.22	110.6	165.9
1.0	2.390			80.01	40.01			
		3.367	61.58			49.23	590.8	554.2
1.0	2.843			176.75	88.37			
		6.110	115.17			137.61	1651.3	1036.5
1.0	3.210			290.85	145.43			
		9.320	175.62			283.04	3396.5	1581.1
.64	2.199			392.81	125.70			
		11.519	217.13			408.74	4904.9	1954.2
.36	1.879			457.37	82.51			
		13.798	241.24			491.25	5895.0	2171.2
1.0	3.663			551.58	275.77			
		16.461	310.29			767.02	8041.0	2742.6
1.0	3.799			692.19	346.10			
		20.260	381.90			1113.12	13357.4	3437.1
.36	1.409			790.19	142.23			
		21.66	408.29			1255.35	15064.2	3674.6
.64	2.513			863.95	276.46			
		24.173	455.66			1531.81	18381.7	4100.9
1.0	3.968			986.12	493.06			
		28.141	530.46			2024.87	24298.4	4774.1
1.0	4.000			1136.32	568.16			
		32.141	605.86			2593.03	31116.4	5452.7
1.0	4.010			1287.31	643.66			
		36.151	681.45			3236.70	38840.4	6133.1

2

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Calculations of Shears and Moments For Drag Fo[rces] From the Airloads

Table II

Condition C2 LAA Symmetrical Maneuver

Col.	1	2	3	4	5	6	7	8
Item	Sta.	C_c	$C_c C$	Sum	Δb Ft	$\frac{1}{2}(C_c C \Delta b)$	$C_c C \Delta b$	Shear
Ref.			② x 5	③$_n$ + ③$_{n+1}$		½ ④ ⑤	Σ ⑥	18.85 ⑦
	132	.0165	.0825					0
				.1045	1.0	.05224		
	120	.0044	.022				.05224	.984
				.053	1.0	.0265		
	108	.0062	.031				.0788	1.489
				.075	1.0	.0375		
	96	.0088	.044				.1163	2.190
				.1035	1.0	.05175		
	84	.0119	.0595				.1680	3.167
				.1265	.64	.04048		
	76.3	.0134	.067				.2085	3.930
				.140	.36	.0252		
	72	.0146	.073				.2337	4.405
				.1545	1.0	.07725		
	60	.0163	.0815				.3109	5.860
				.1640	1.0	.0845		
	48	.0175	.0875				.3954	7.453
				.1775	.36	.03195		
	43.6	.0180	.090				.4274	8.056
				.181	.64	.05792		
	36	.0182	.091				.4853	9.118
				.1845	1.0	.09225		
	24	.0187	.0935				.5775	10.88
				.1865	1.0	.09325		
	12	.0186	.093				.6707	12.64
				.1855	1.0	.09275		
	0	.0185	.0925				.7635	14.39

...d Moments For Drag Forces — Wing Loads

2

Table IX

(5) Δb Ft.	(6) $\Delta(C_c C \Delta b)$ $\frac{1}{2}$(4)(5)	(7) $C_c C \Delta b$ Σ(6)	(8) Shear 18.85(7) 0	(9) Sum $(8)_n + (8)_{n+1}$	(10) ΔM $\frac{1}{2}$(5)(9)	(11) Moment Ft-Lb Σ(10) 0	(12) Moment In-Lb 12×(11) 0
1.0	.03225			.9849	.4924		
		.08995	.9849			.492	5.90
1.0	.0265			2.470	1.235		
		.0788	1.485			1.727	20.72
1.0	.0375			3.675	1.837		
		.1163	2.190			3.564	42.77
1.0	.05175			5.357	2.678		
		.1680	3.167			6.242	74.90
.64	.04048			7.097	2.271		
		.2085	3.930			8.513	102.16
.36	.0252			8.335	1.500		
		.2337	4.405			10.013	120.16
1.0	.07725			10.265	5.132		
		.3109	5.860			15.145	181.74
1.0	.08415			13.313	6.656		
		.3954	7.453			21.801	261.61
.36	.03145			15.507	2.792		
		.4274	8.056			24.593	295.12
.64	.05792			17.204	5.505		
		.4853	9.148			30.098	361.18
1.0	.09225			20.034	10.017		
		.5775	10.886			40.115	481.38
1.0	.09325			23.527	11.763		
		.6707	12.643			51.878	622.54
1.0	.09275			27.035	13.517		
		.7635	14.392			65.395	784.74

WING LOADS

Calculation of Shears, Moments and Torques from the Inertia Loads

The condition solution for $n_z = 2.5$ was obtained using the wing unit solution, $n_z = 1$, shears and moments and the appropriate load factor. Since the angle of wing incidence is small, resolution of inertia loads in the chordwise direction is small and is conservatively neglected. The centroid of the inertia load was assumed to be located at reference axis. Therefore, no torque results from inertia loads.

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WING LOADS

Calculation Shears And Moments From The Inertia Loads.

Limit Loads
Table X

$w = .1775$ #/in

Tip Sta. x

Assuming Uniform Distribution of Wing Weight $w = \frac{46.9}{264} = .1775$ #/in.

Cond.	$n_z = 1$ Unit Solution					$n_z = 2.5$ Econ	
Item	Running Load	Distance From Tip		Shear Lbs	Moment In-Lbs	Shear Lbs	Moment In-Lbs
Col.	1	2	3	4	5	6	7
Ref				1 x 2	1 x 3	2.5 x 4	2.5 x 5
Sta.	w	x	x^2	wx	$\frac{wx^2}{2}$		
132	-.1775	0	0	0	0	0	0
120		12	144	-2.13	-13	-5.3	-32.5
108		24	576	-4.26	-51	-10.7	-123
96		36	1296	-6.39	-115	-16.0	-288
84		48	2304	-8.52	-204	-21.3	-510
76.3		55.7	3102	-9.87	-275	-24.7	-685
72		60	3600	-10.65	-319	-26.6	-798
60		72	5184	-12.78	-460	-31.9	-1150
48		84	7056	-14.91	-626	-37.2	-1567
43.6		88.4	7814	-15.70	-697	-39.2	-1742
36		96	9216	-17.04	-818	-42.5	-2045
24		108	11,664	-19.17	-1035	-47.9	-2590
12		120	14,400	-21.30	-1280	-53.2	-3200
0	-.1775	132	17,424	-23.43	-1546	-58.6	-3860

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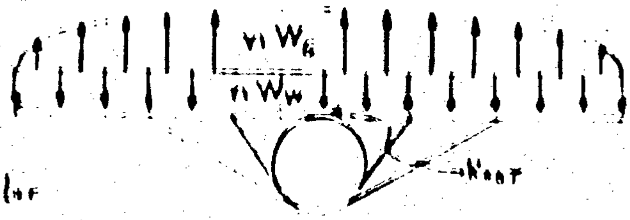
WING LOADS

Calculation of Shears, Moments, and Torques From The Brace Cable Loads

The cable loads under conditions of flight are determined from values measured under static test conditions. The figure below shows that the wing inertia load when compared to the airload is:

(a) opposite in direction but multiplied by n in flight
(b) in same direction and constant in a static test.

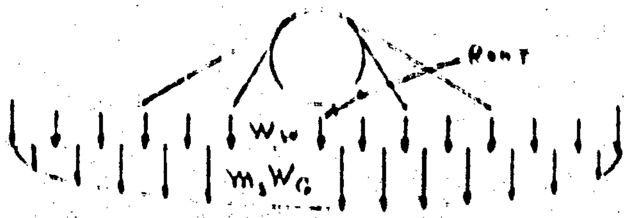


n = Flight Load Factor

(a) Flight Load Condition

W_G = Gross Wt.
= 550 LB
W_W = Wing Wt.
≅ 50 LB

m_s = Static Test Load Factor



(b) Static Test Load Condition
Aircraft is Inverted

Figure Relation Between Airload and Wing Inertia Load For Flight and Static Test Conditions

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Wing Geometry

Lower Wing Brace Cable Coordinates In Wing Reference System

Points	X	Y	Z
A	14.0	76.3	4.4
B	34.1	76.3	3.6
C	11.0	43.6	4.4
D	19.0	8.0	40.0
E	50.8	4.0	25.9

Calculation of Direction Cosines For Cables

Member	X	Y	Z	L	$\frac{X}{L}$	$\frac{Y}{L}$	$\frac{Z}{L}$	Check
AD	5.0	68.3	35.6	77.18	.06478	.88494	.46125	1.0001
CD	8.0	35.6	35.6	51.12	.15650	.69643	.70435	.9999
BE	16.7	72.3	22.3	77.48	.21553	.93314	.28781	1.0000

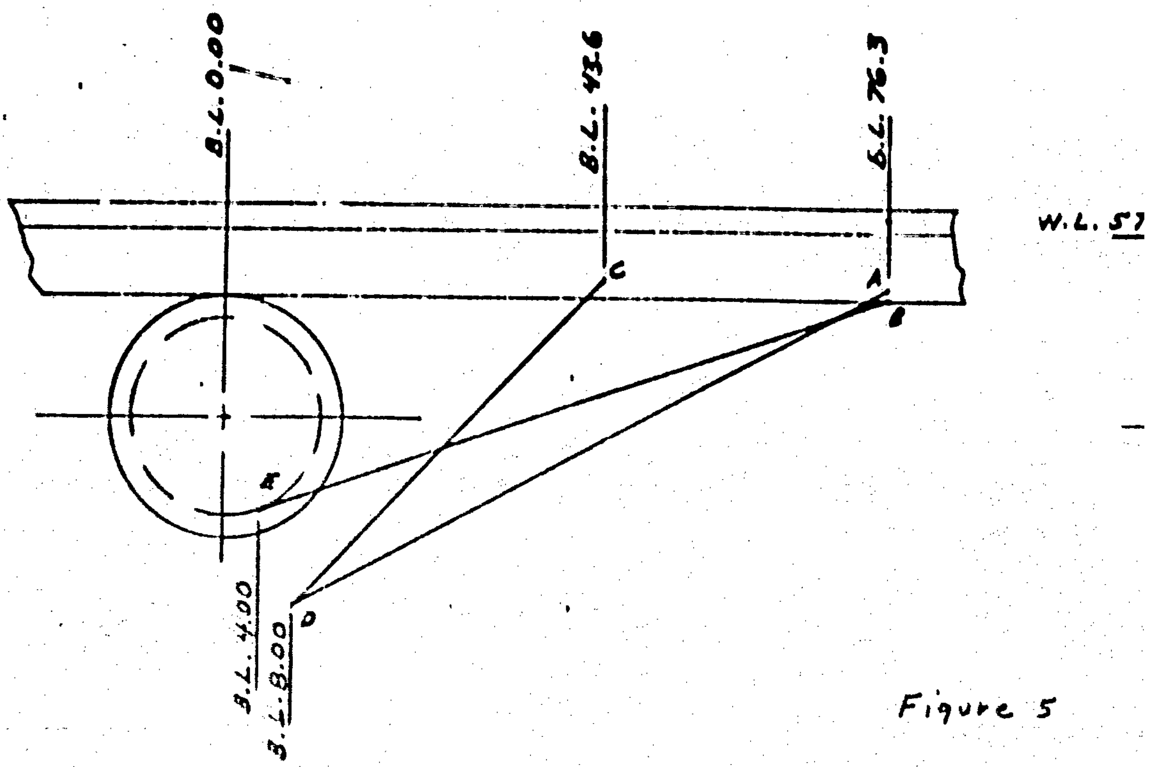

Figure 5

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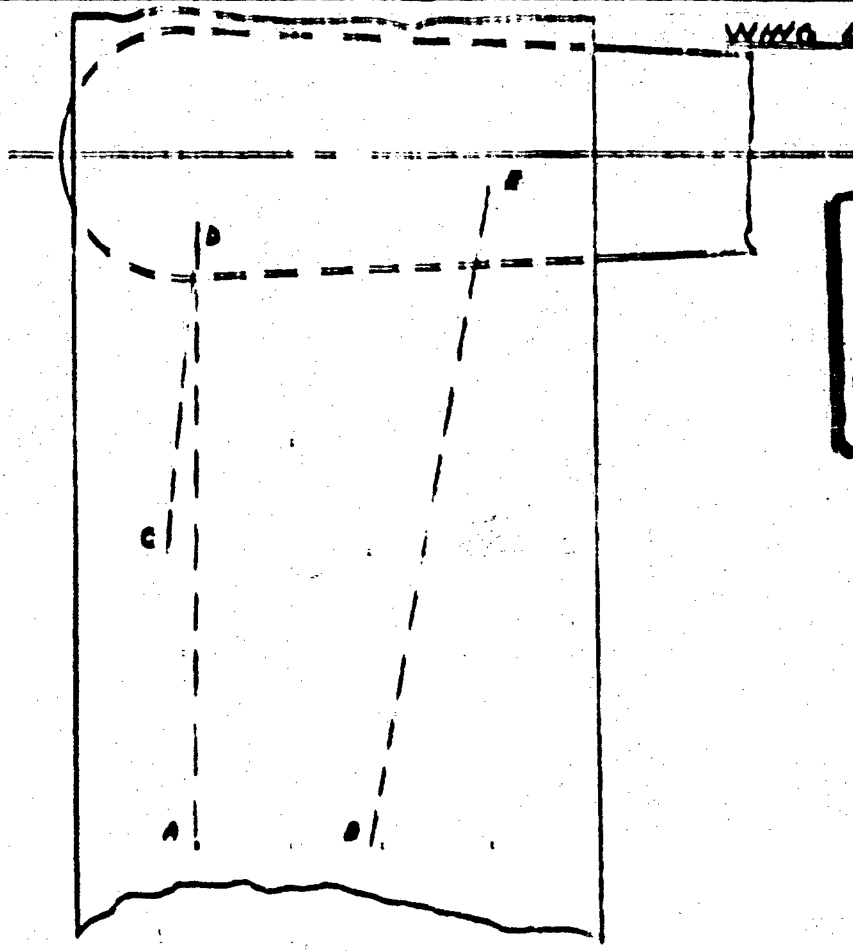
WING LOADS

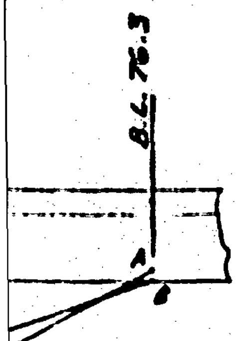
...ates

E
4.4
3.6
4.2
40.0
25.9

For Cables

Y/L	B/L	Check
.88444	.46125	1.0001
.69643	.70425	.9999
.93314	.28781	1.0000





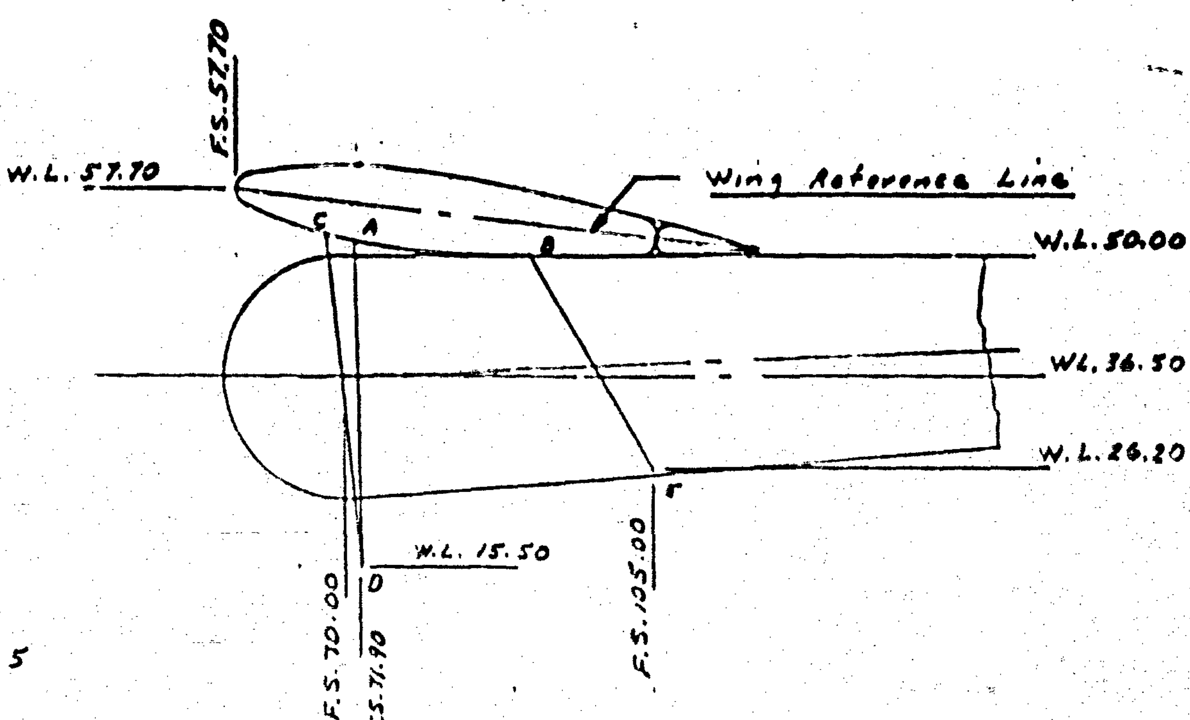


Figure 5

WING LOADS

The first table below gives the root moments due to airloads and wing inertia loads for various values of load factor. The next table gives cable loads measured under static test conditions for various load factors.

Table XI

①	②	③	④	⑤	⑥
Load Factor n or n_g	Root Moments In. Lb.				
	Air load Distribution (a)	Wing Weight (a)	Wing Weight (b)	Net Root Moment (a)	Net Root Moment (b)
Ref				② − ③	② + ④
1	15,500	1545	1545	13,955	17,045
2	31,000	3090	1545	27,910	32,545
2.5	38,800	3860	1545	34,940	40,345
3	46,500	4635	1545	41,865	48,045
4	62,000	6180	1545	55,820	63,545
5	77,500	7725	1545	69,775	79,045

Table XII

①	⑥	⑦	⑧	⑨
Load Factor n_g Ref.	Net Root Moment (b)	Inboard Cable Load CD lb.	Forward Outboard Cable Load AD lb.	Aft Outboard Cable Load BG lb.
1	17,045	140	218	85
2	32,545	210	415	214
3	48,045	325	600	352
4	63,545	400	820	500

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WING LOADS.

Cable loads under flight conditions are determined from Figure [illegible] in which the load factor n and M_s are plotted vs net root moment. On the same figure the static test cable loads are plotted vs static test net root moments. By assuming that the cable loads under flight conditions are dependent upon the net root moment it is possible to read flight cable loads from flight load factors n. The table below gives the cable loads corrected to flight load factors n.

Table XIII

Flight Load Factor n	Cable Loads, Lb CD Inboard	AD Fwd Outbd.	BE Aft Outbd.
1	120	140	60
2	200	350	170
2.5	240	440	220
3	280	540	300
4	360	720	420
4.375	390	790	475

← Used as an example

Table XIV

Cable	Load Lb	Direction Cosines x	y	z	Components, Lb. x	y	z
CD	240	.1563	.69613	.70035	37.5	167.1	168
AD	440	.06478	.88494	.46125	28.5	389.4	203
BE	220	.21553	.97314	.28781	49.6	214.6	66

Component	Sum of Components at Sta 43.6 CD	at Sta 76.3 AD + BE
x	37.5	78.1
y	167.1	604.0
z	168	269

The cable loads, components, and sum of components at Stations 43.6 and 76.3 are given to left and above.

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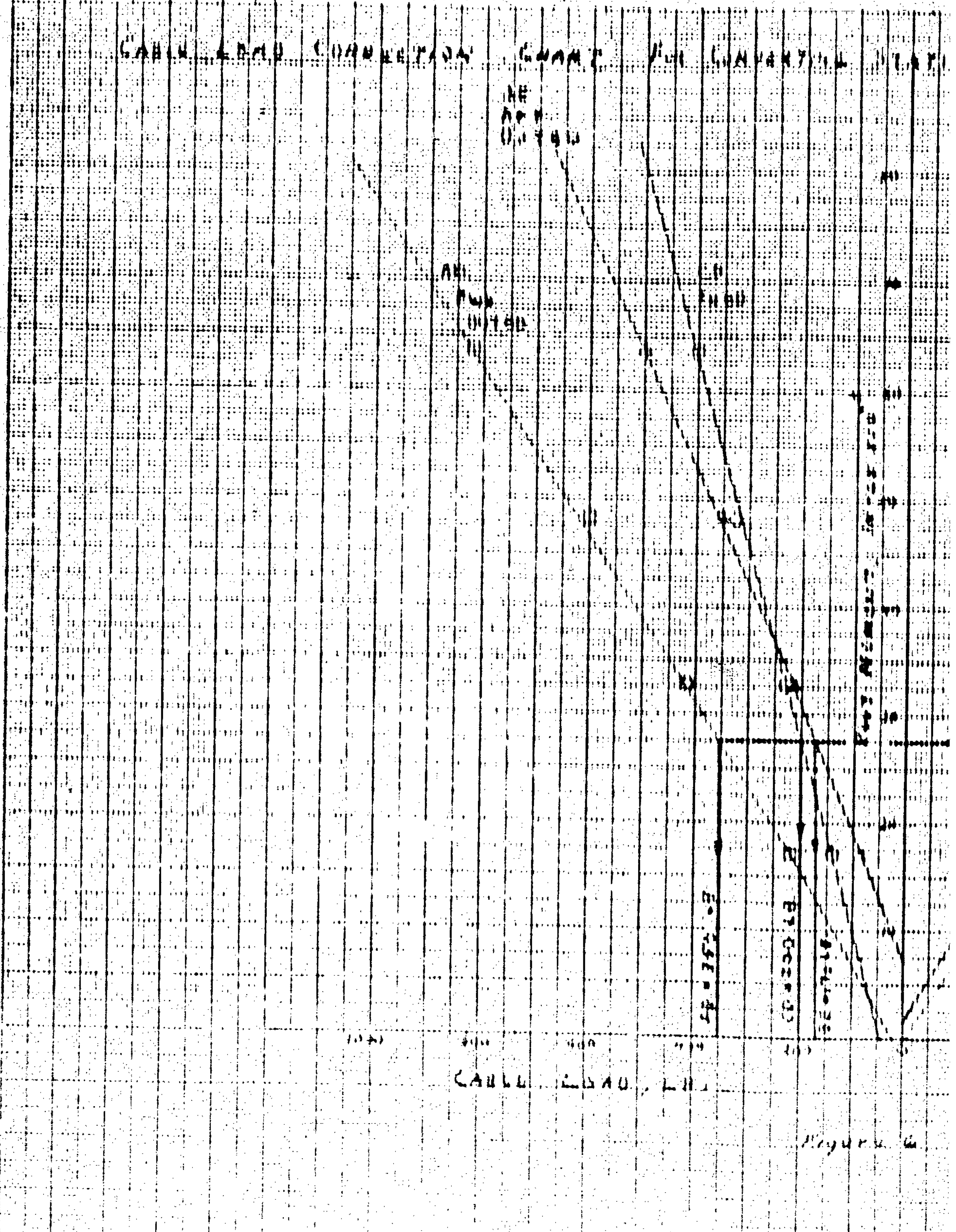


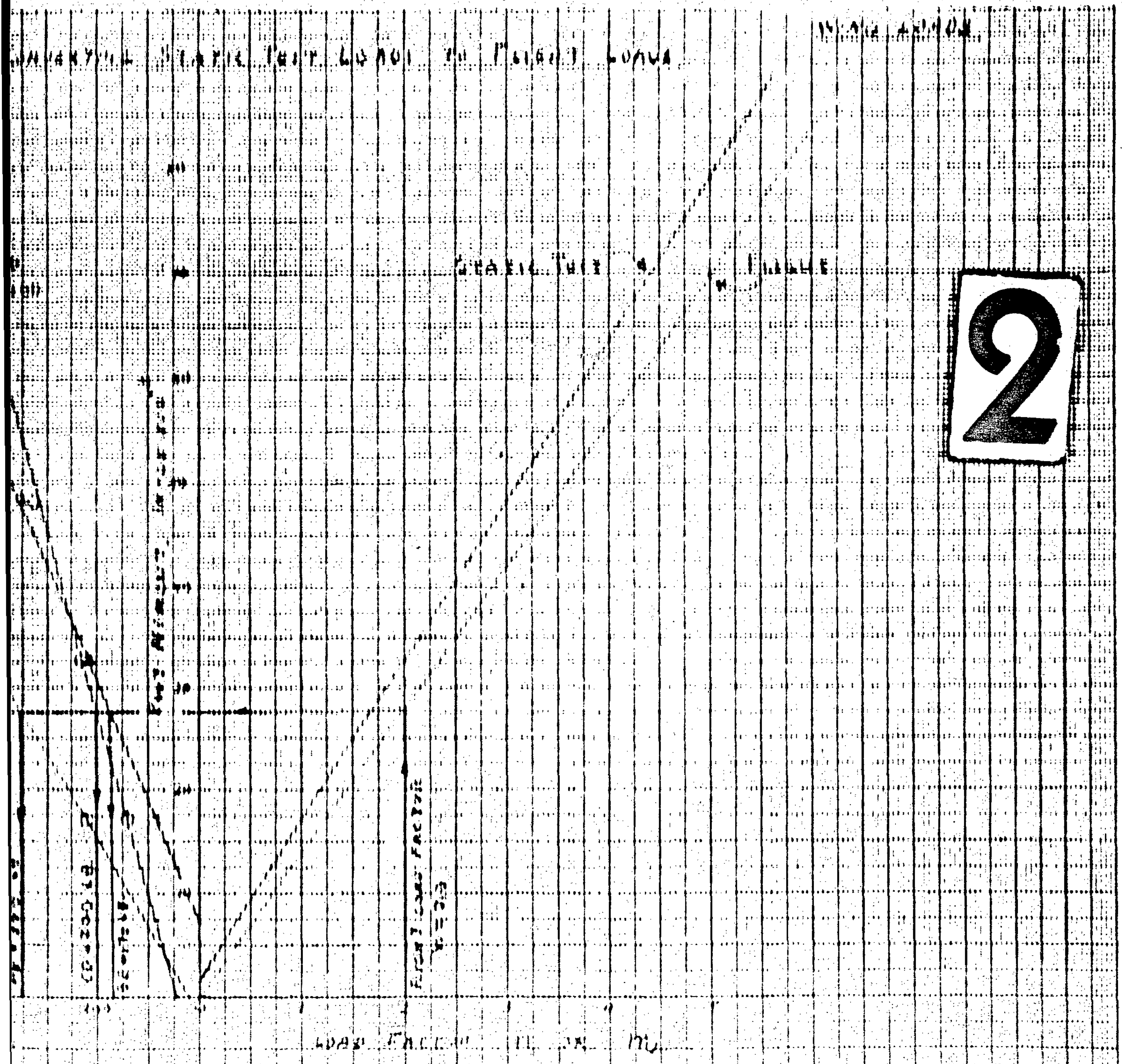
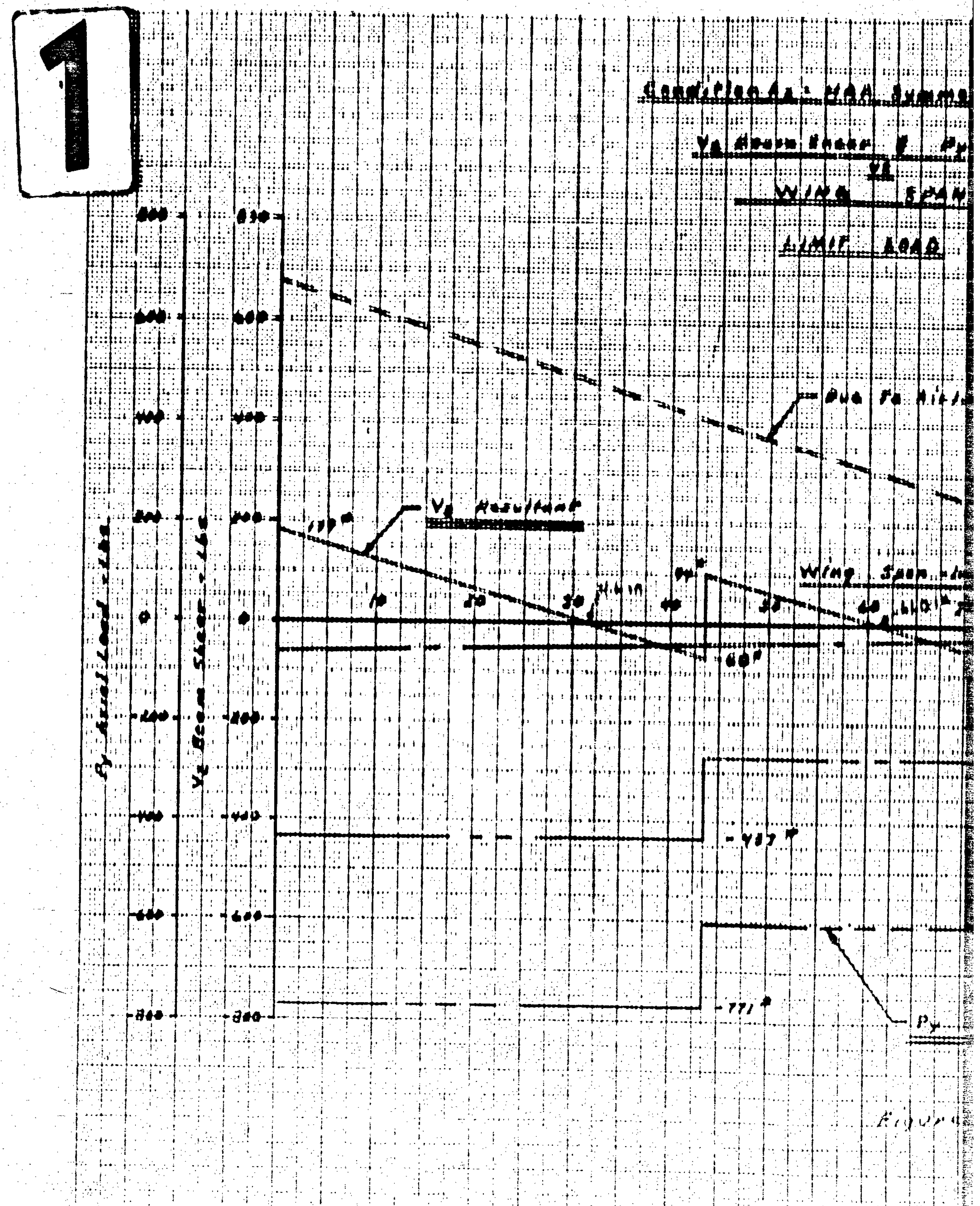
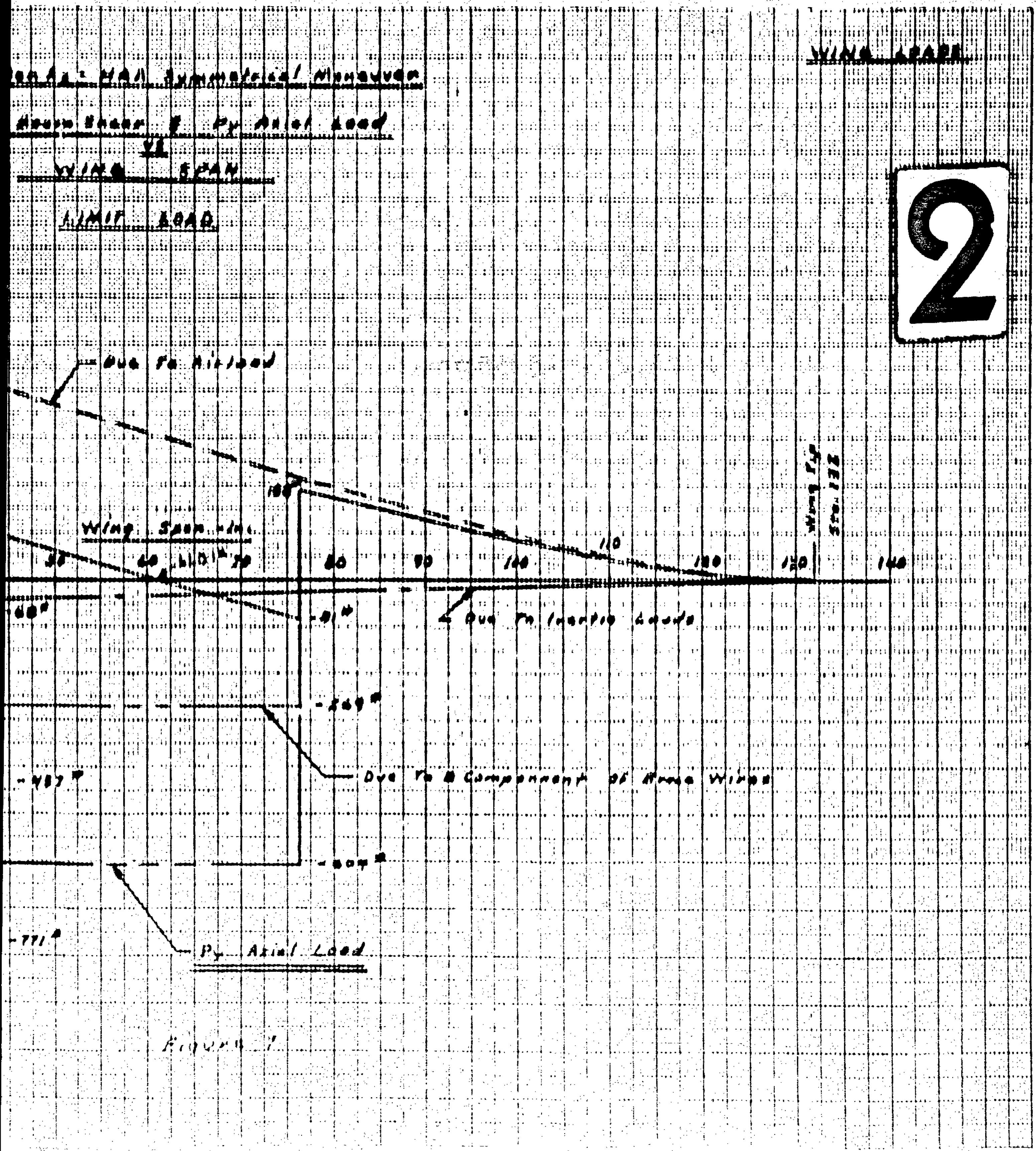
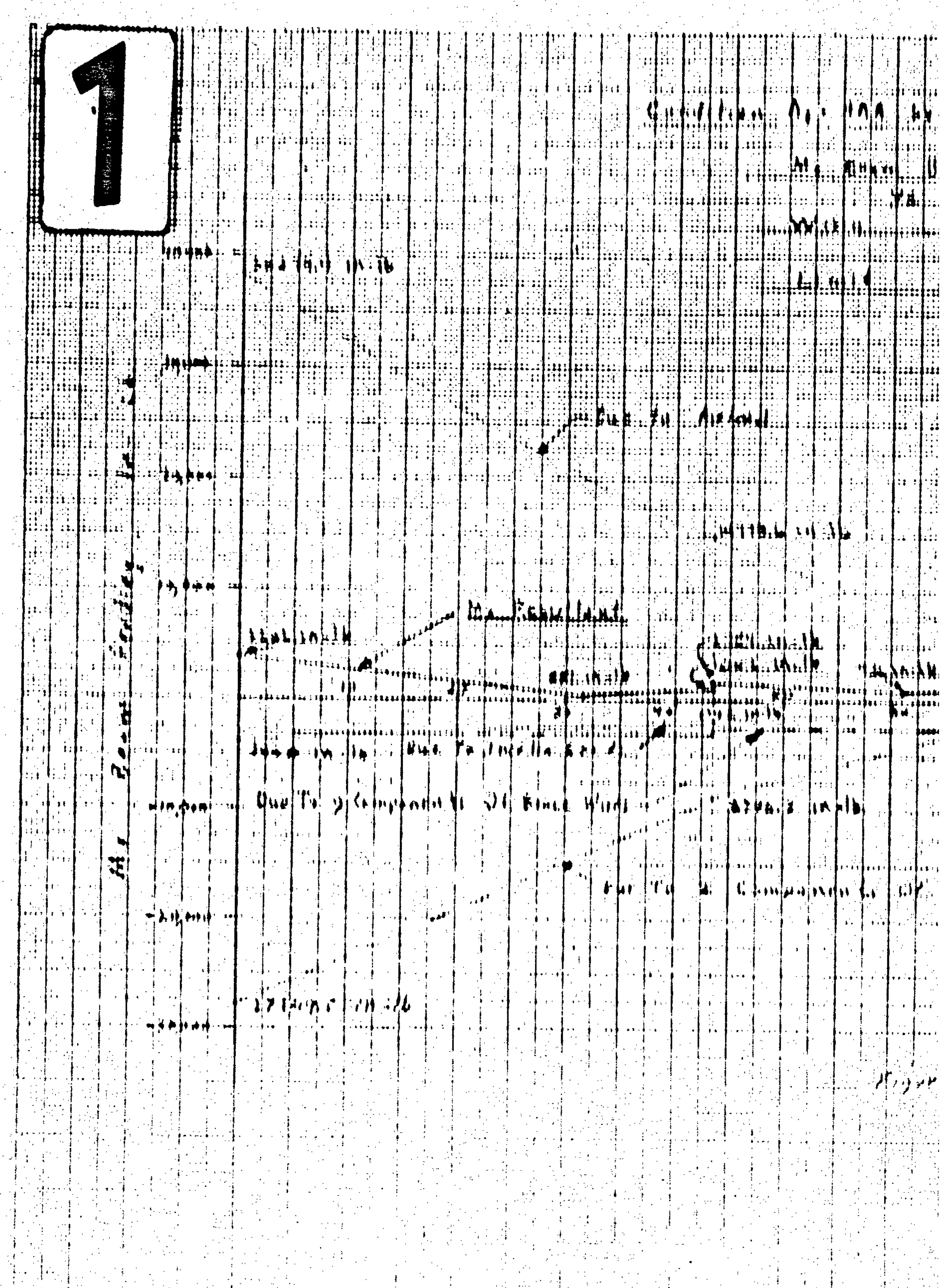


Figure 6.



LIMIT LOAD
Vz Resultant
Py Axial Load - Lbs
Vz Beam Shear - Lbs
Wing Span
-787#
-771#
Py


Hoop Shear & Py Axial Load
vs
Wing Span
Limit Load
Due To Airload
Wing Span-in.
80
90
100
110
120
130
140
Wing Tip
Sta. 132
Due To Inertia Loads
-269#
-457#
Due To Component of Flying Wires
-809#
-771#
Py Axial Load



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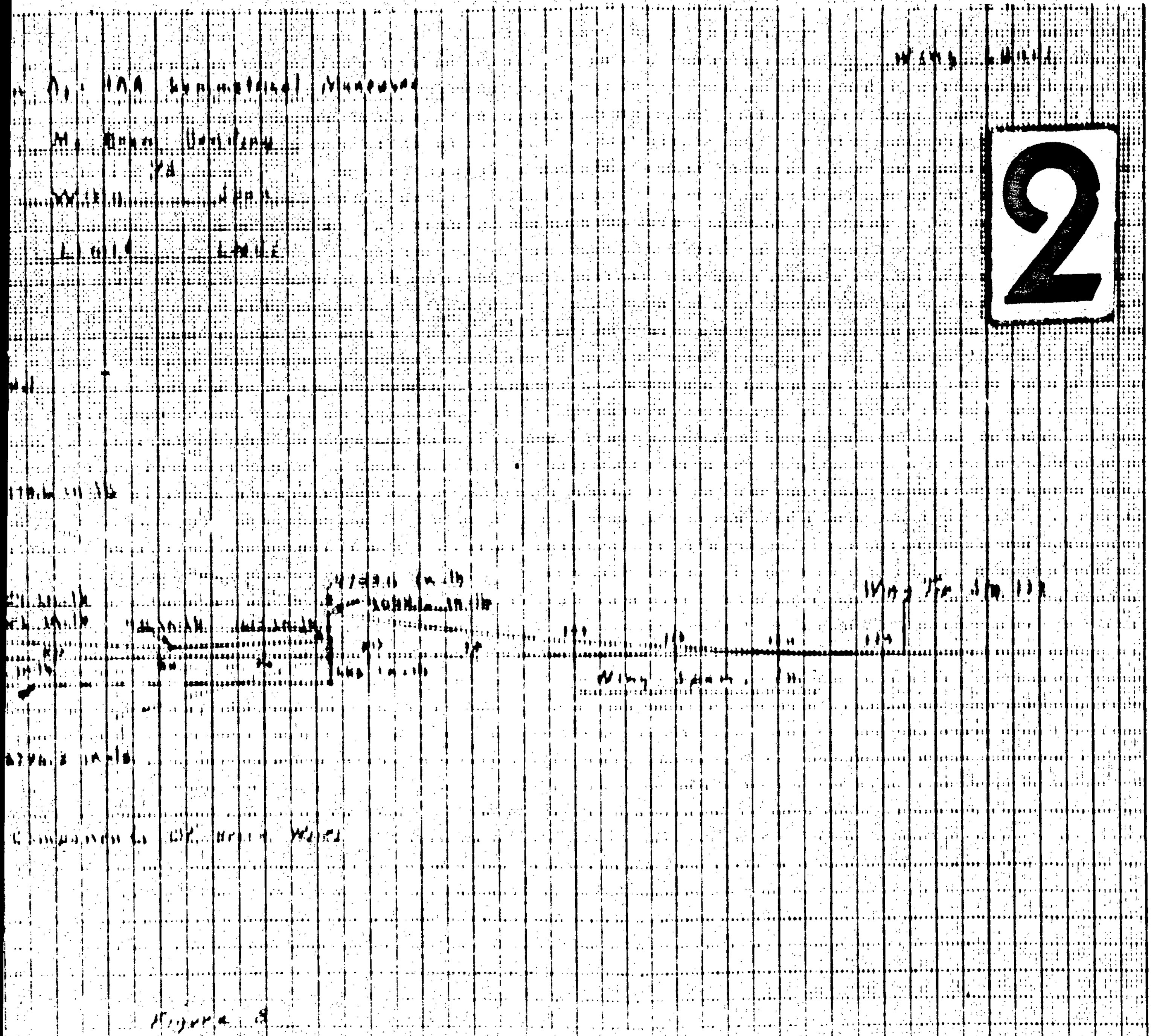
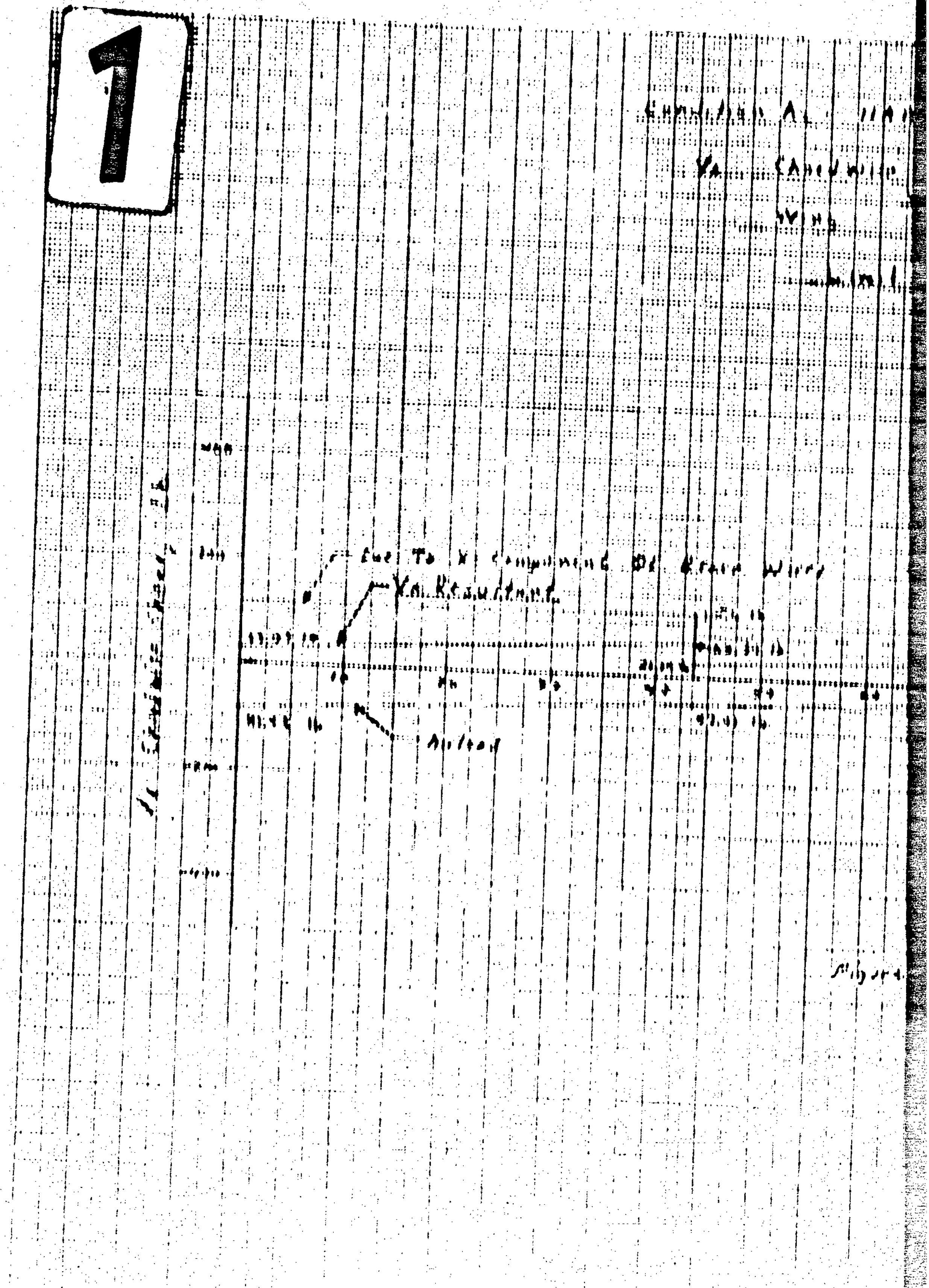


Figure 8



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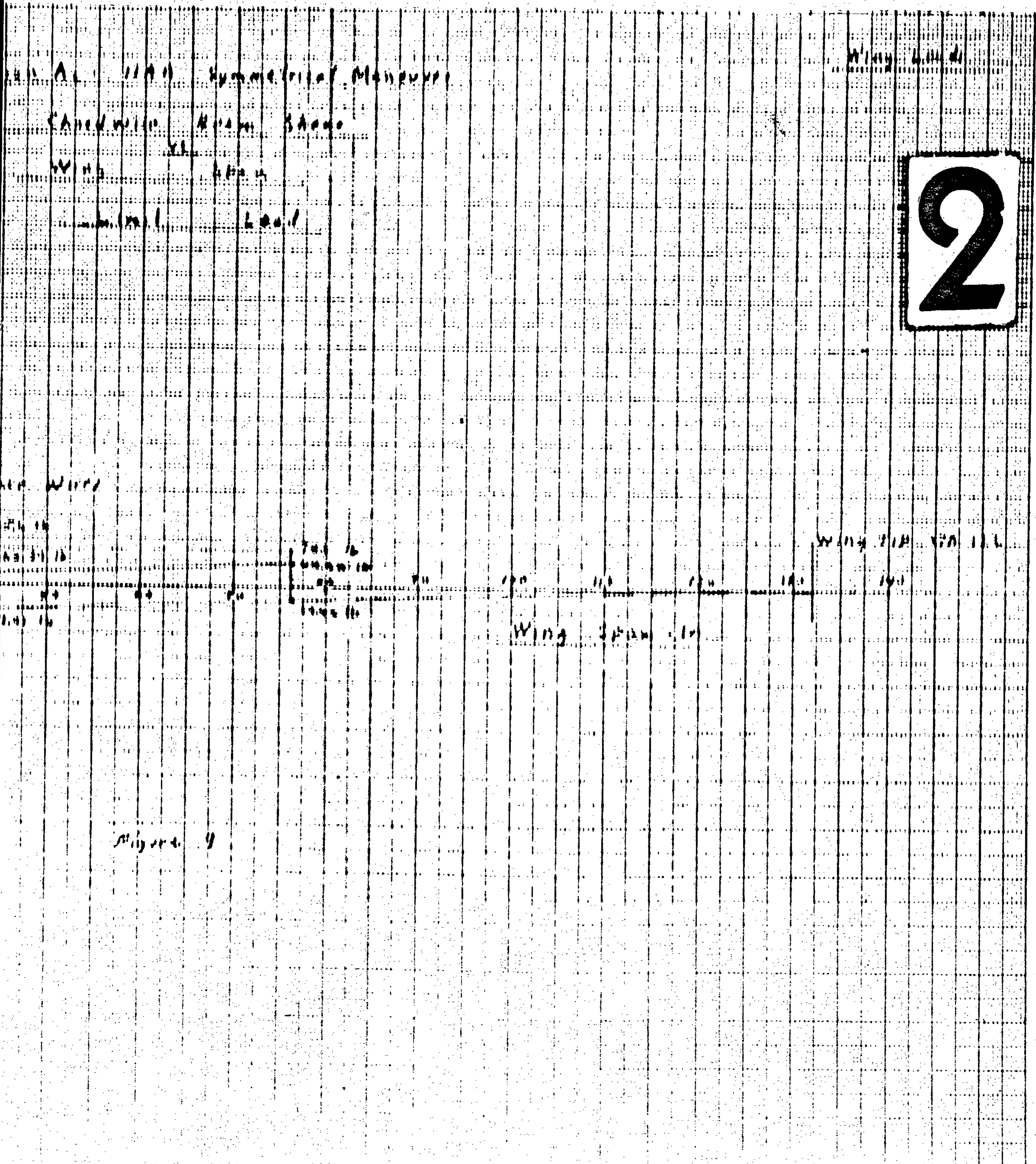
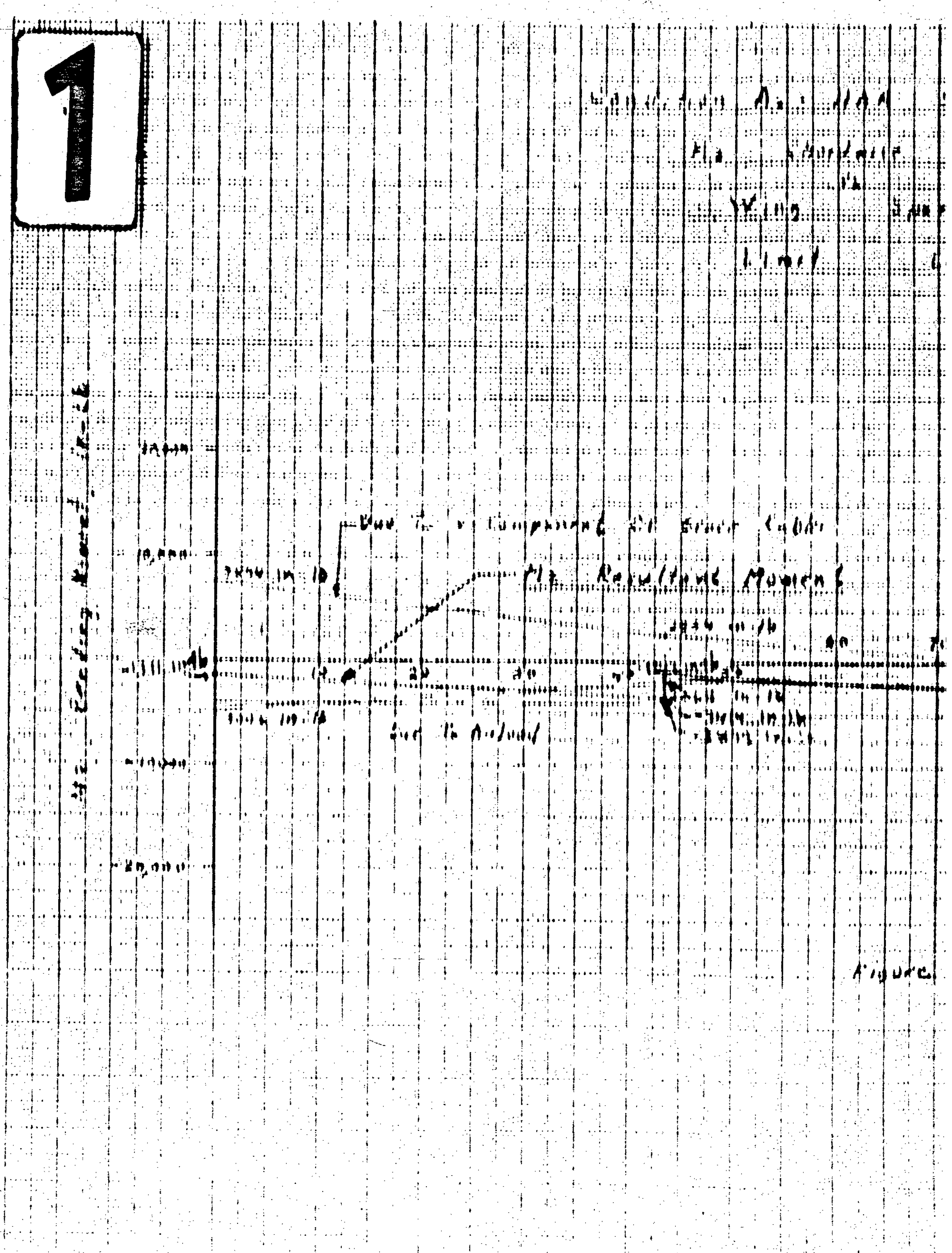


Figure 4



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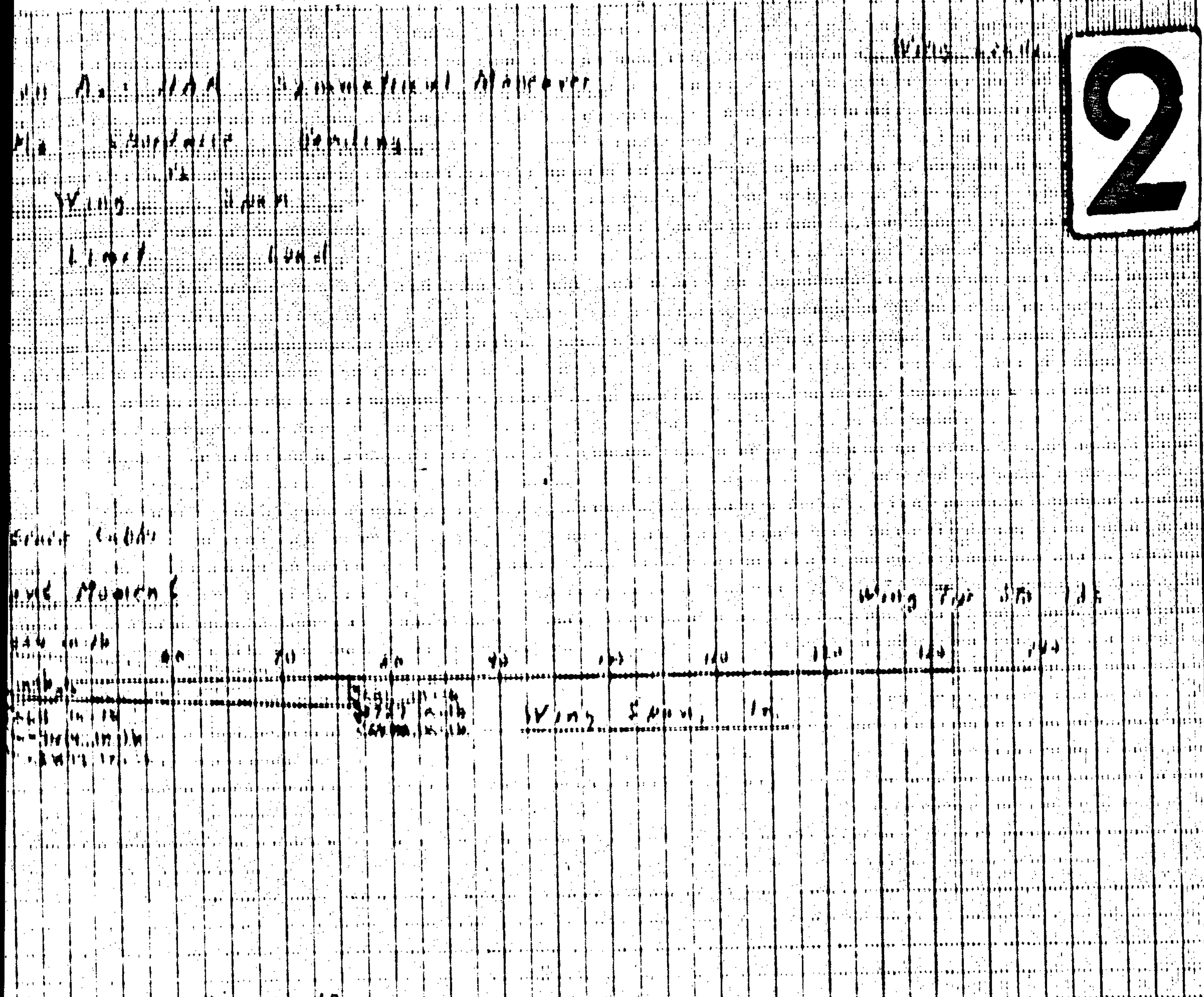
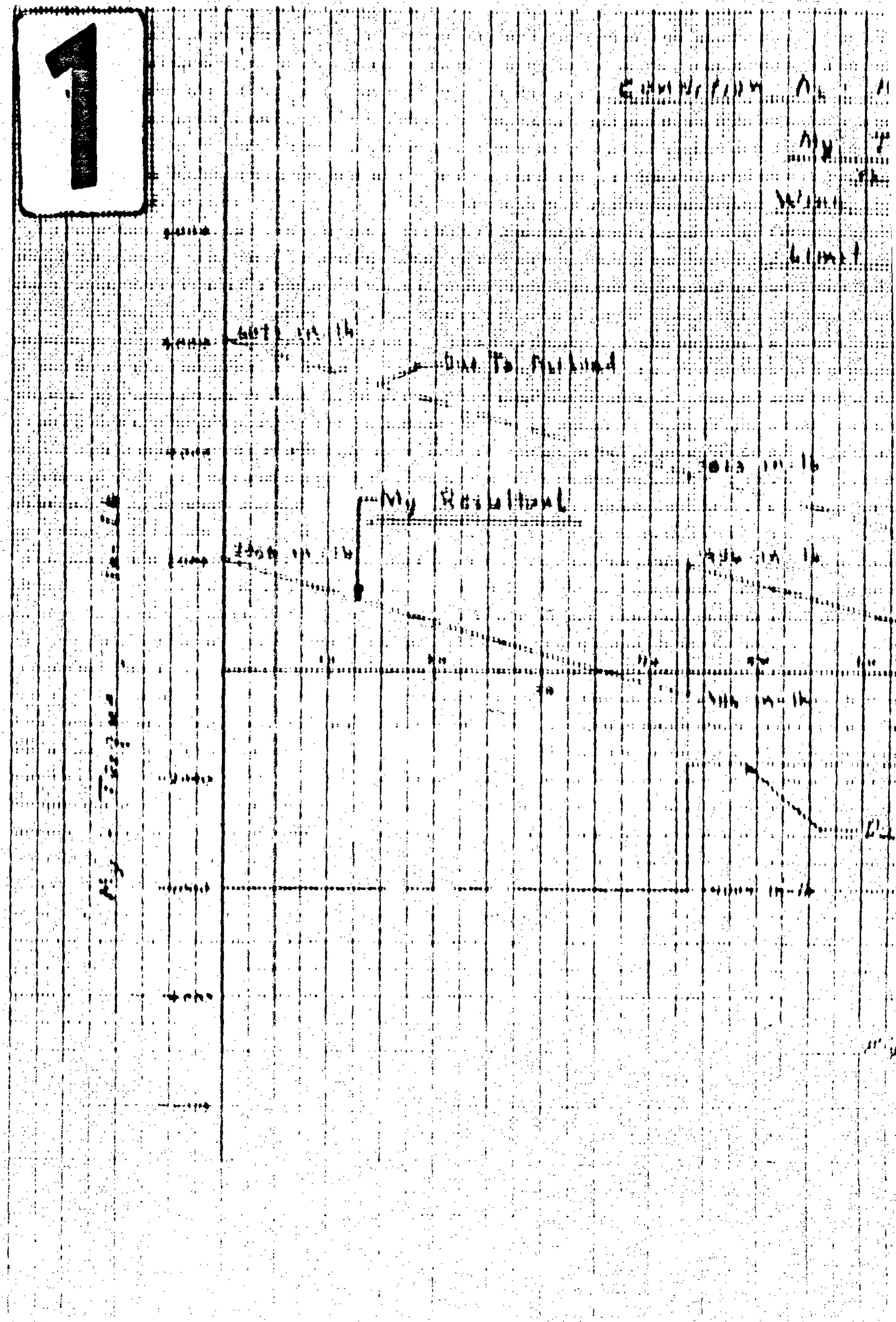
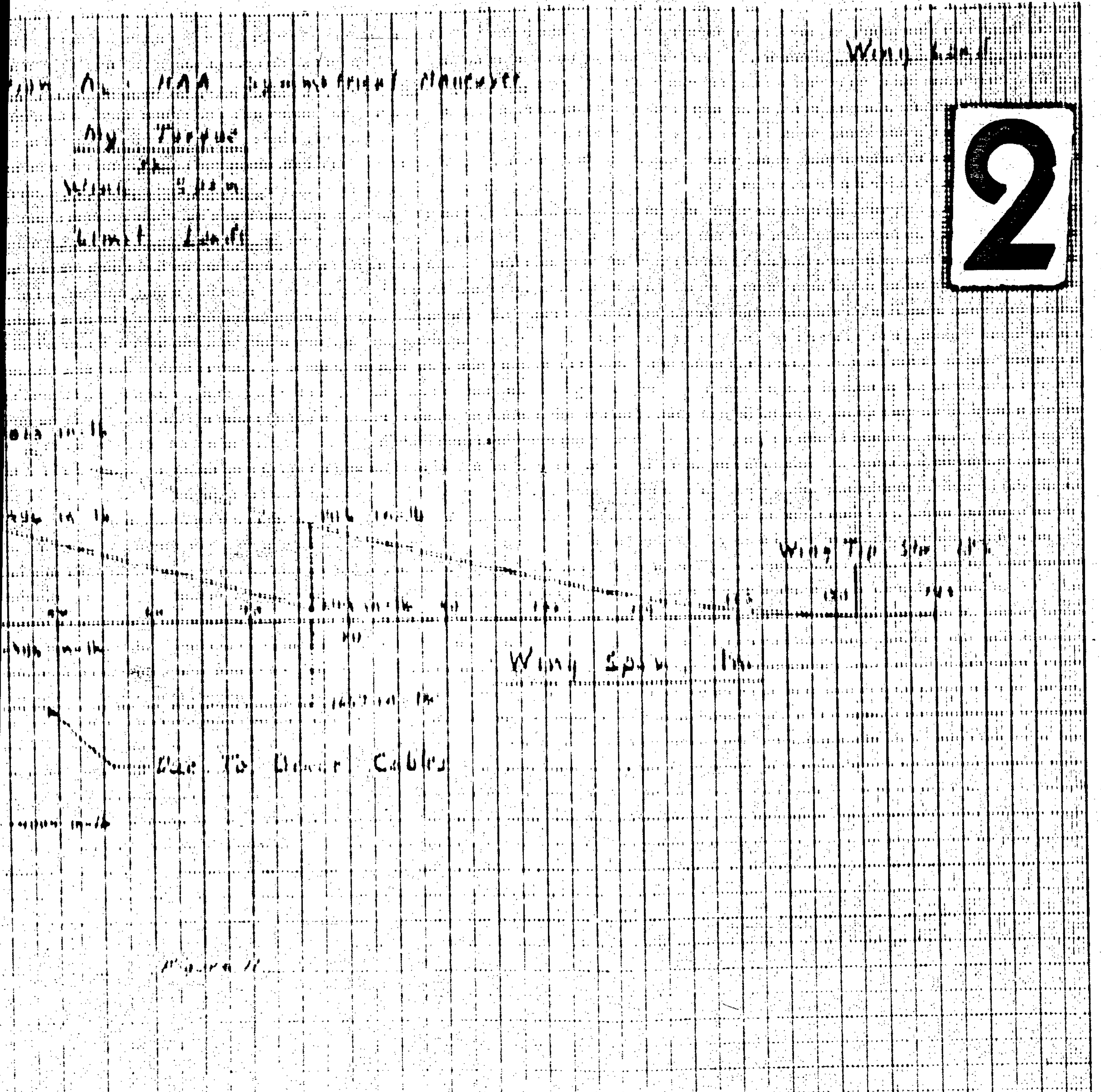


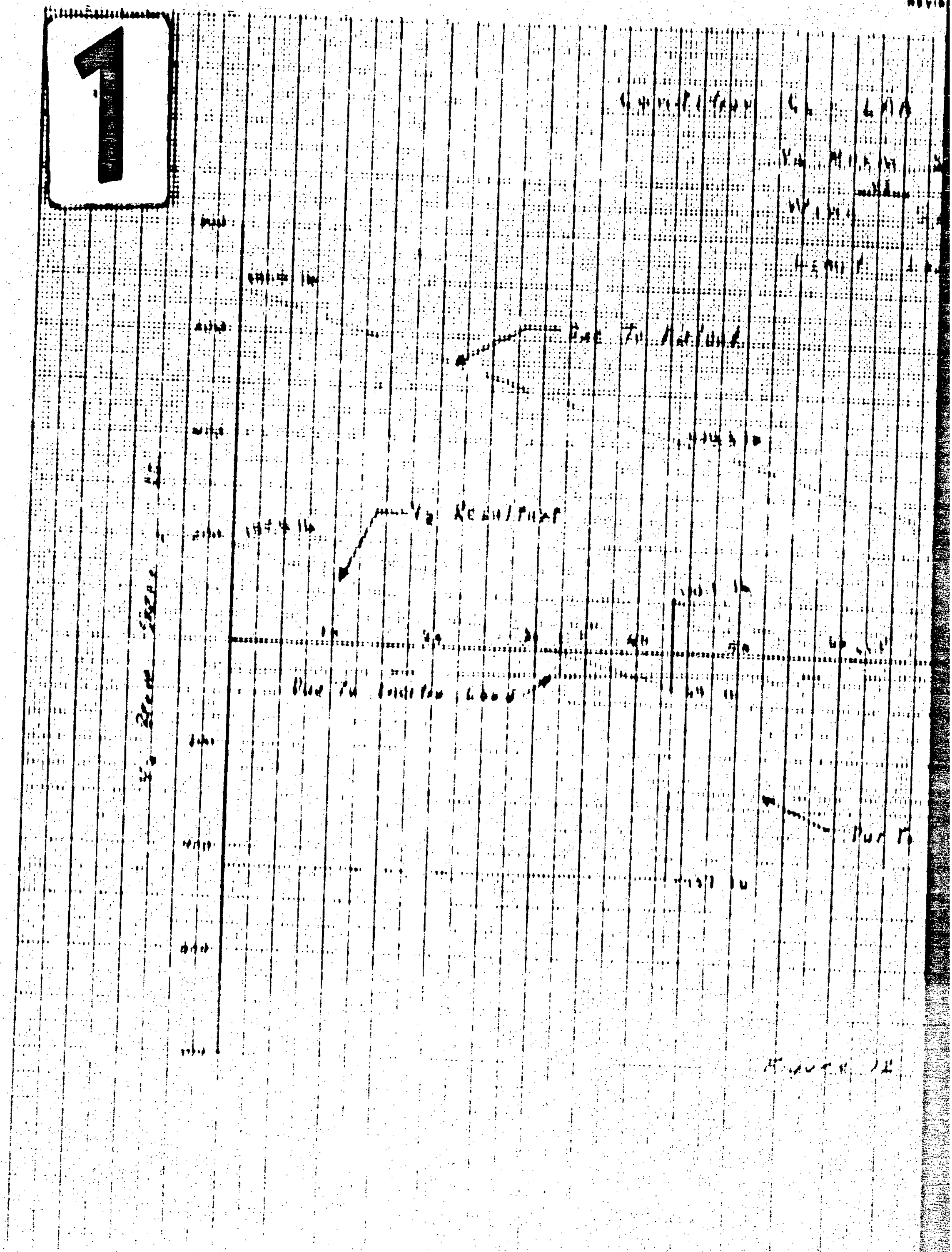
Figure 10


My Residual



Wing Load
My Torque
Wing Span
Limit Loads
Wing Tip Sta.
Wing Span, In.
Due To Decoy Cables

2



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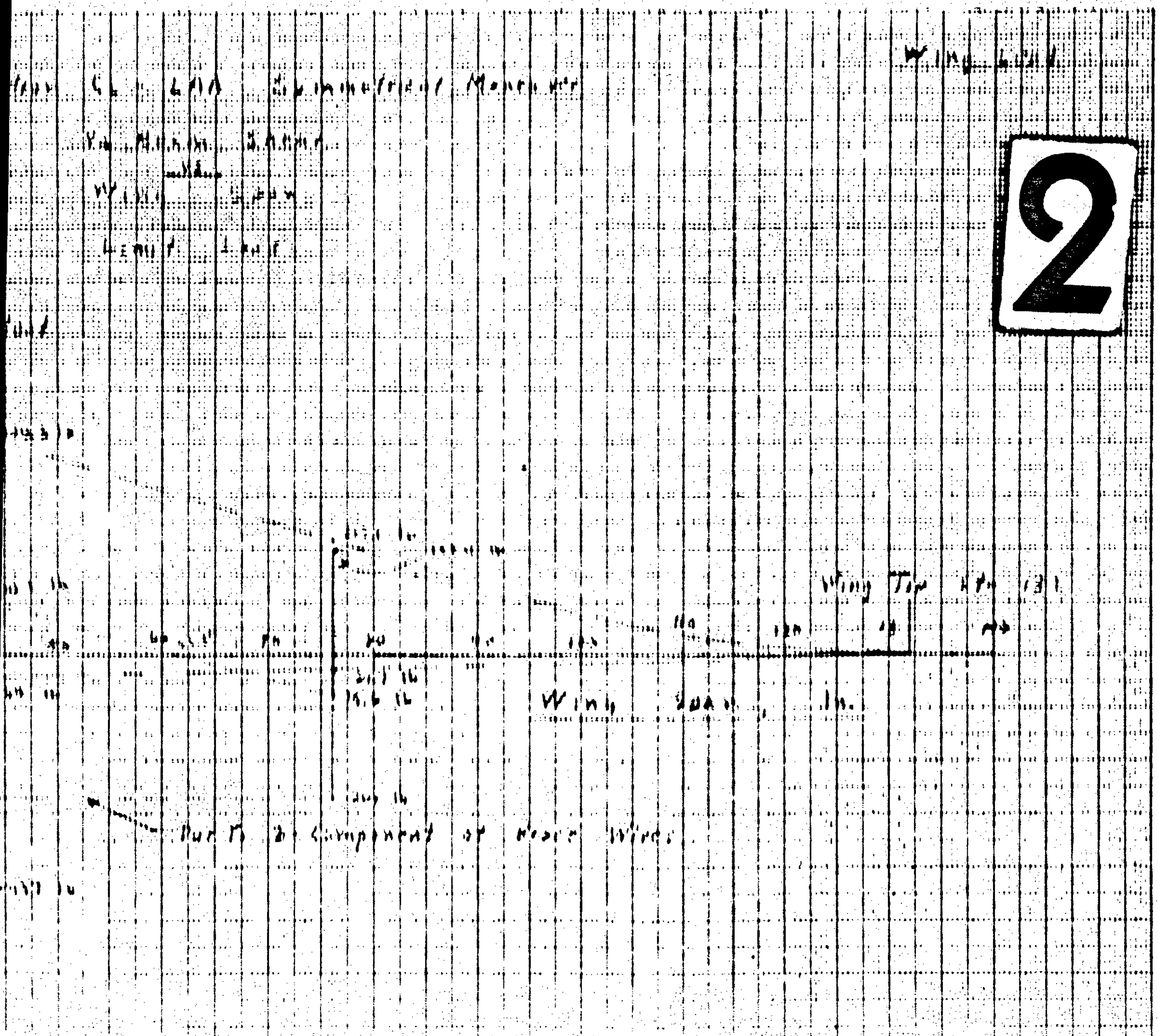
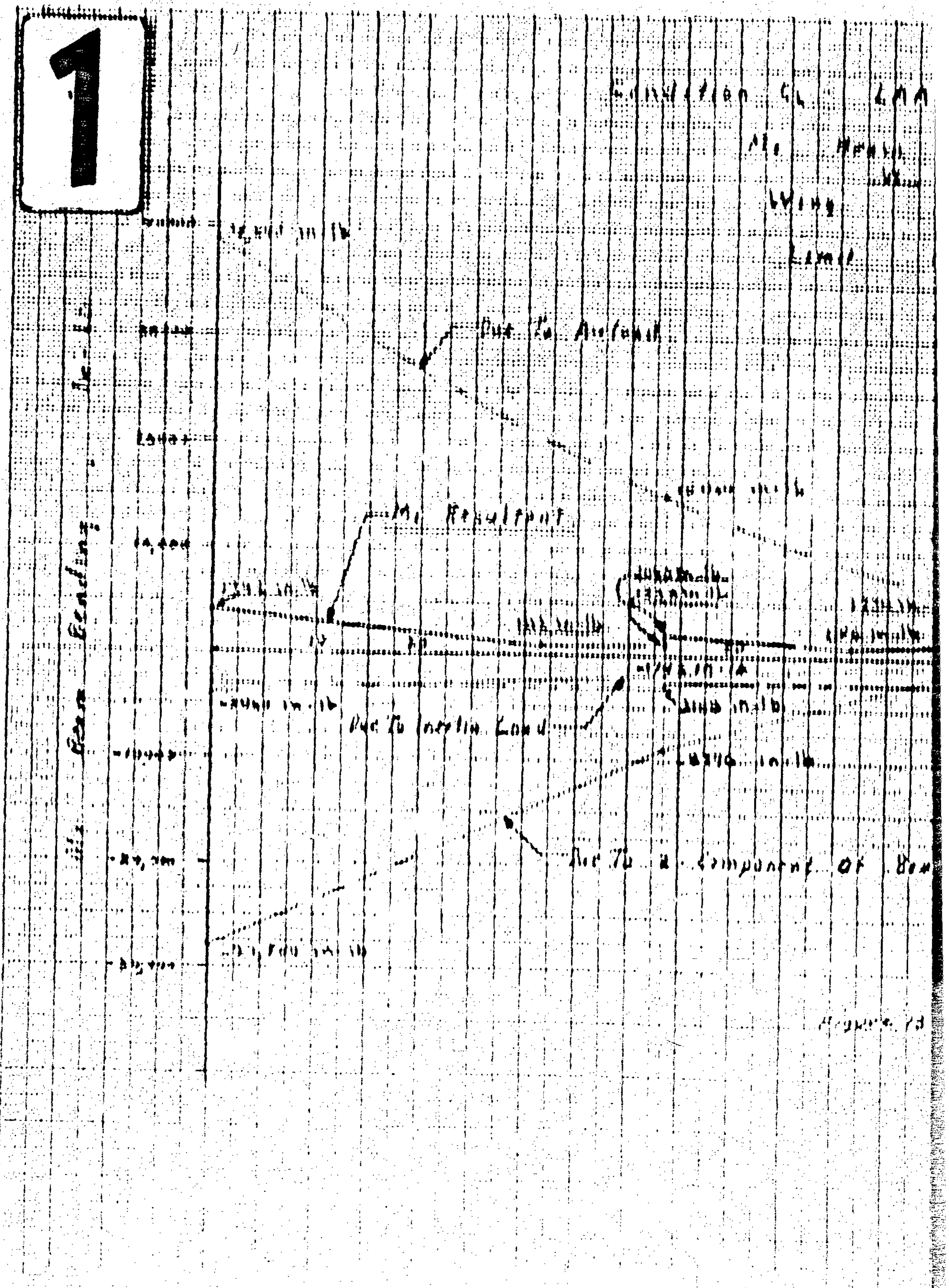


Figure 12


Due To Airload
Due To Inertia Load
Beam Bending

Cond. C4 — LAA Symmetrical Maneuver,
Max. Heavy Bending
Wing Spar
Limit Load

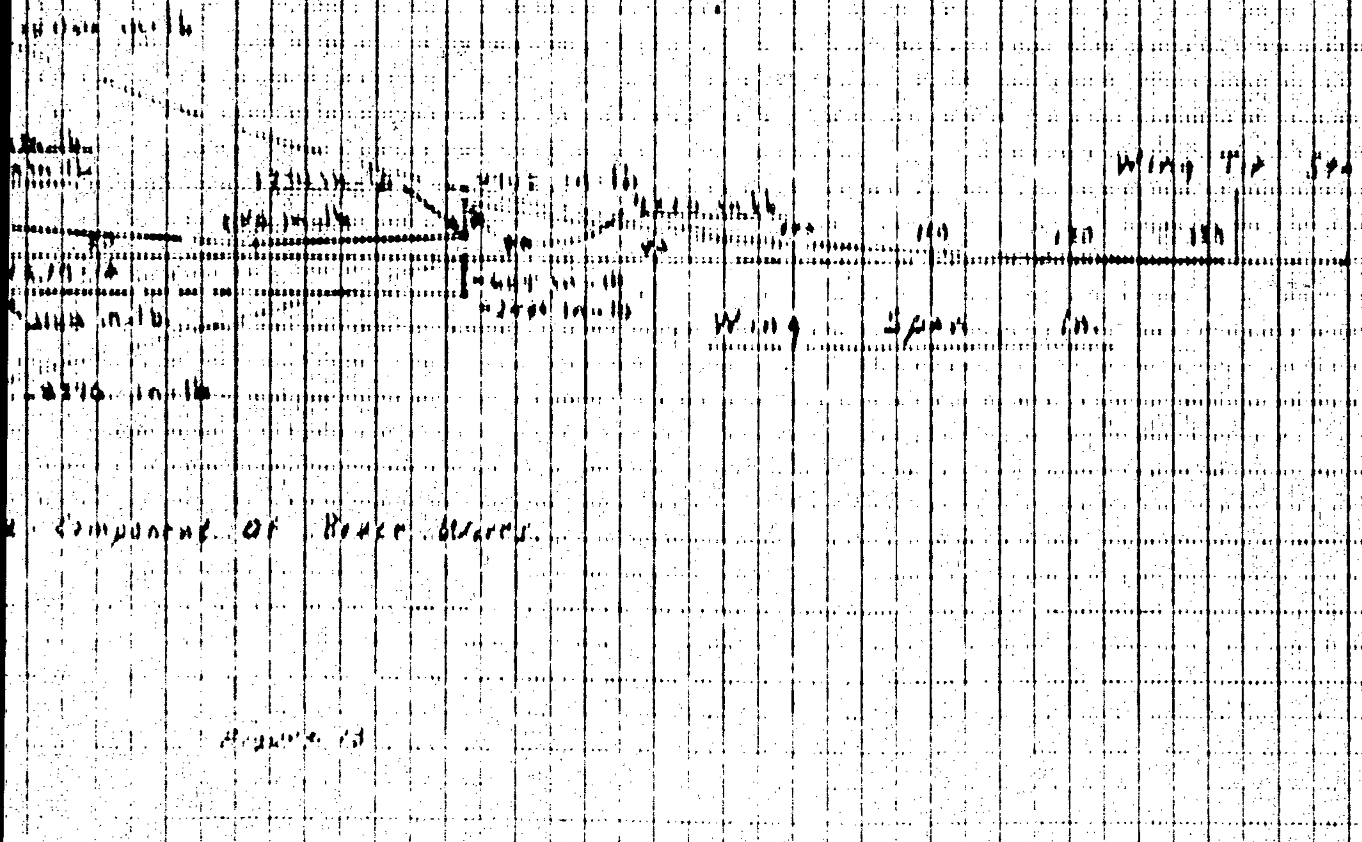
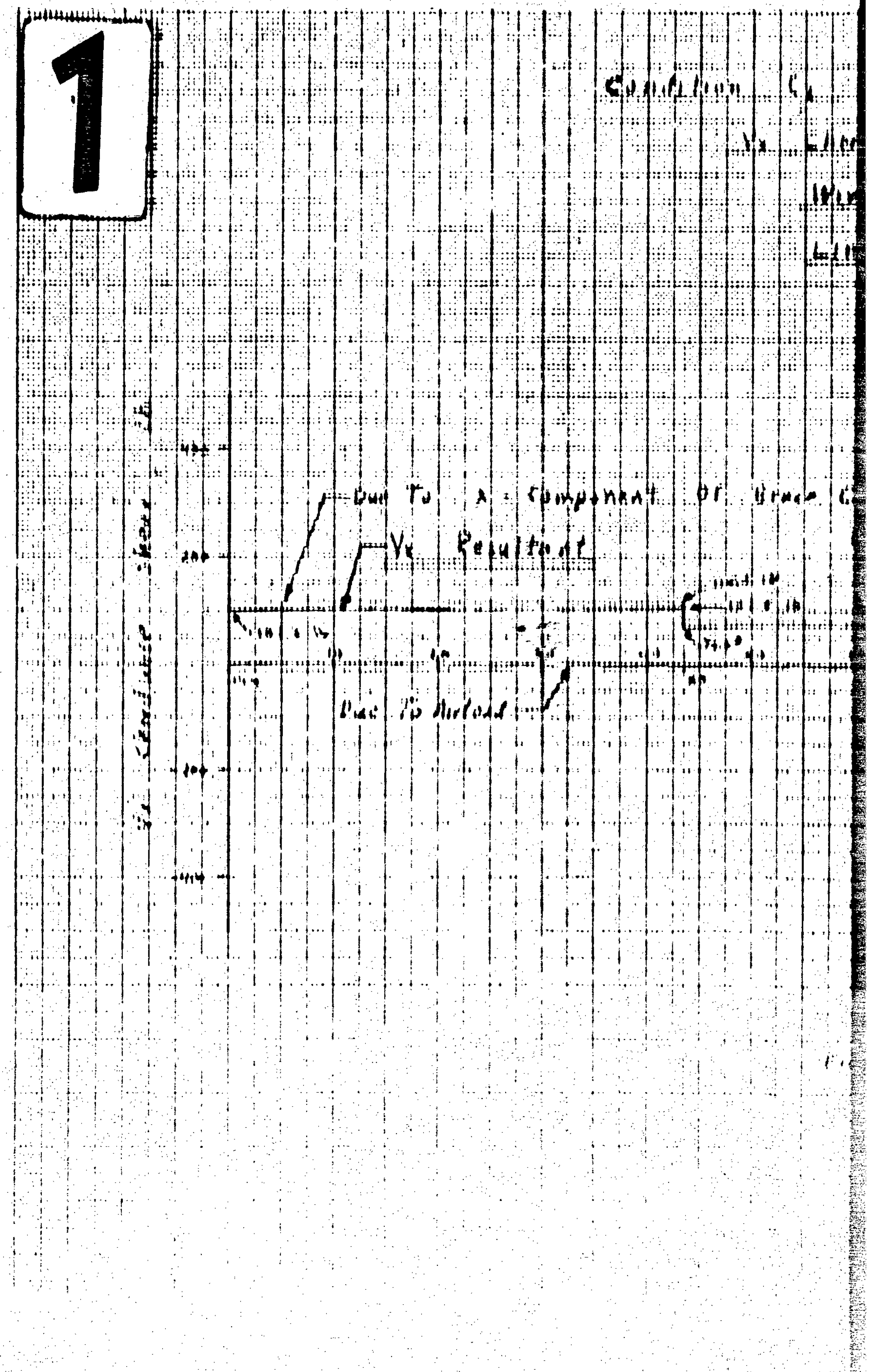


Figure 13



Due To x Component Of Brace
Vx Resultant
Due To Airload

Wing Loads

... C_L ... Symmetrical Maneuver

Vs ... Shear ...

Wing Sta.

Limit Load

For Brace Cables

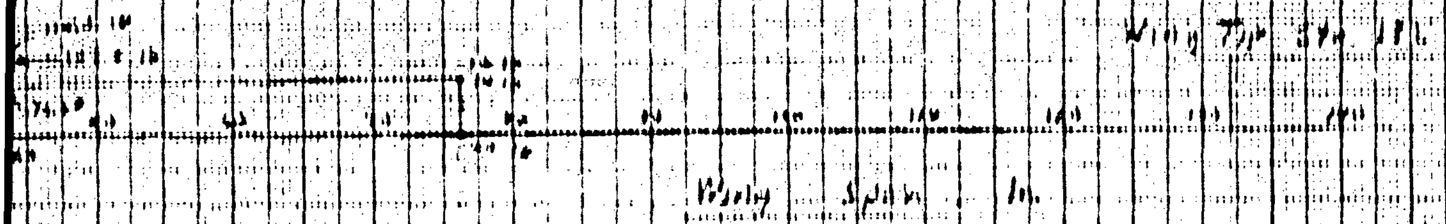
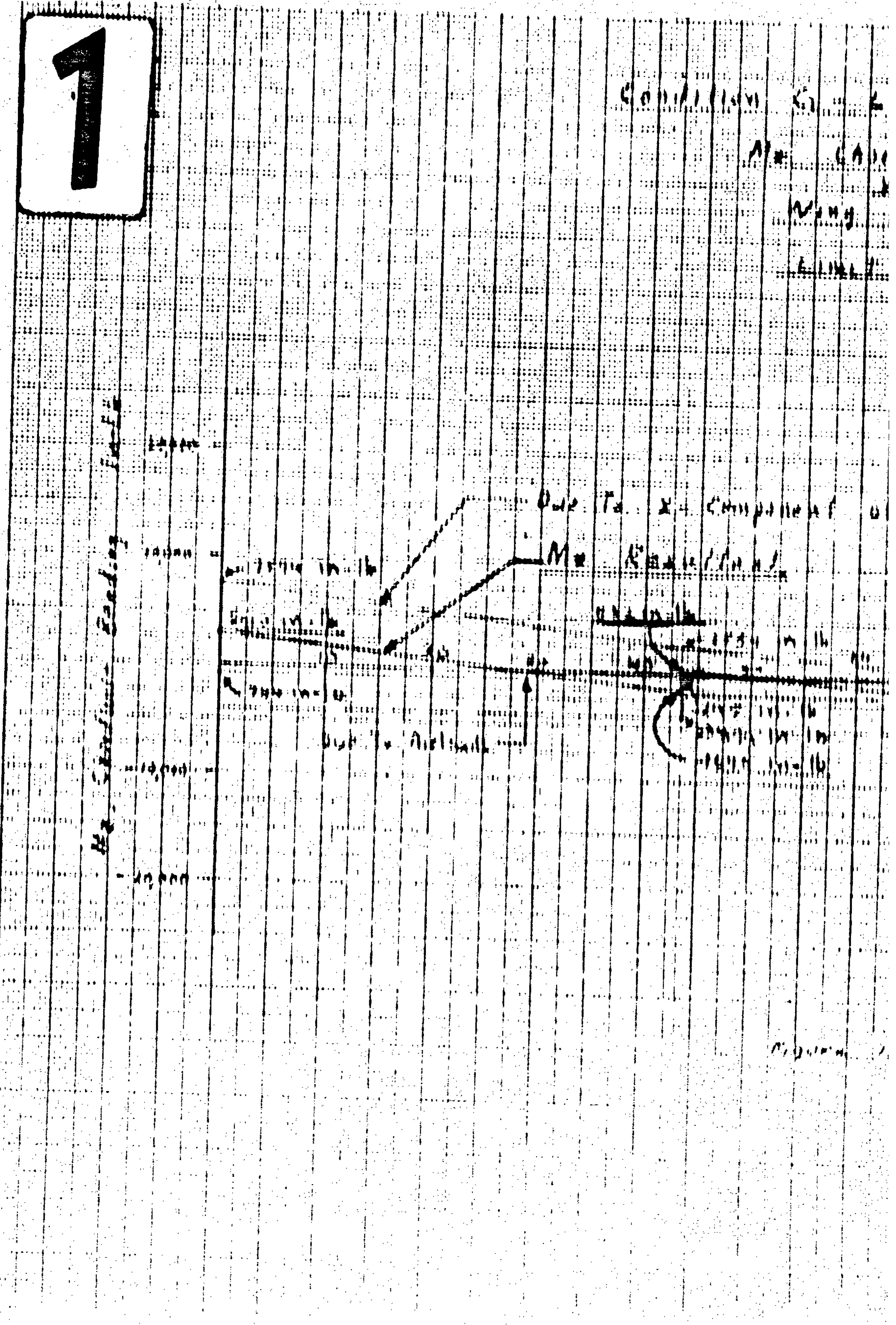


Figure 24


Due To x-Component o
Mx Resultant
Due to Actual

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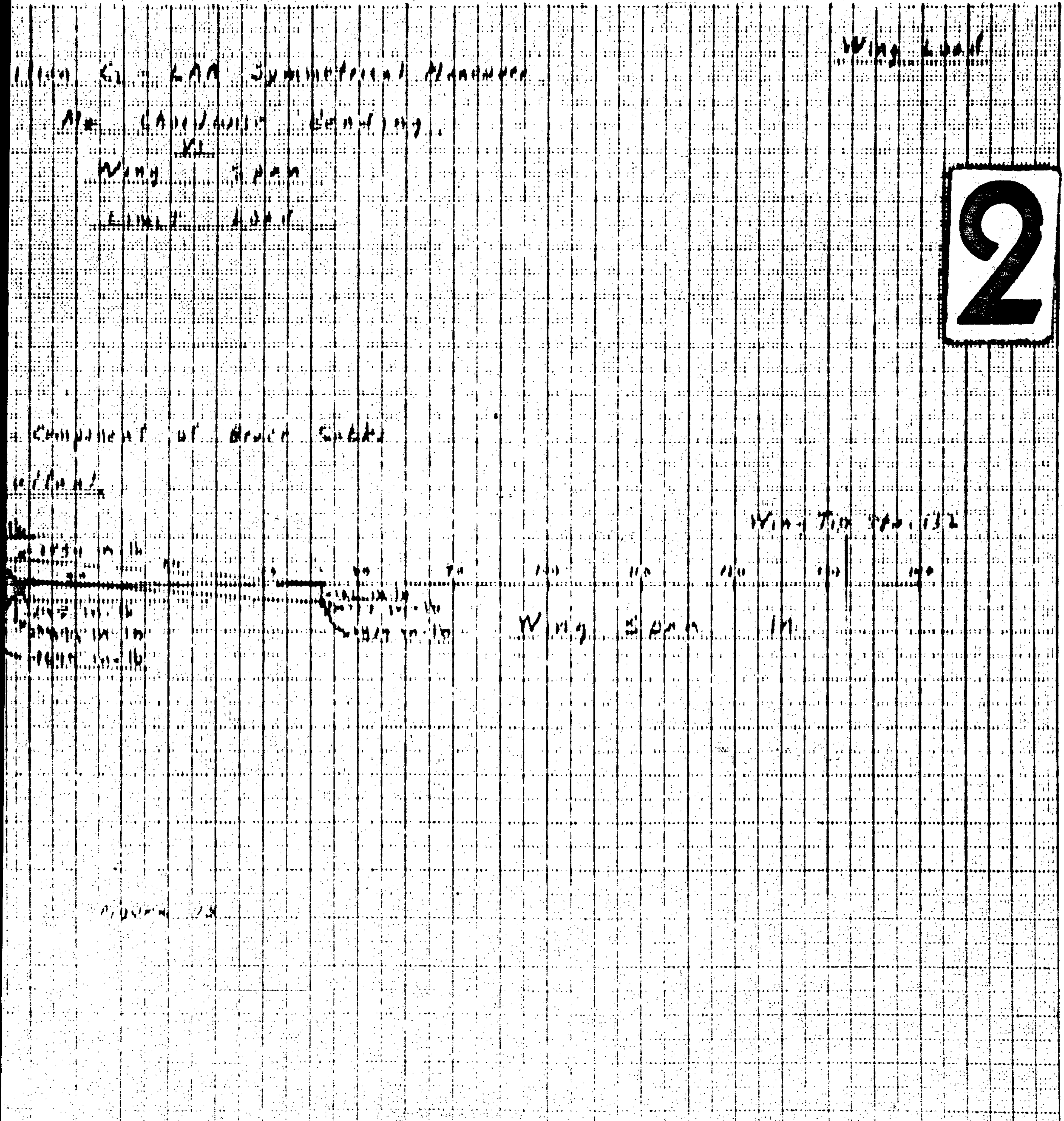

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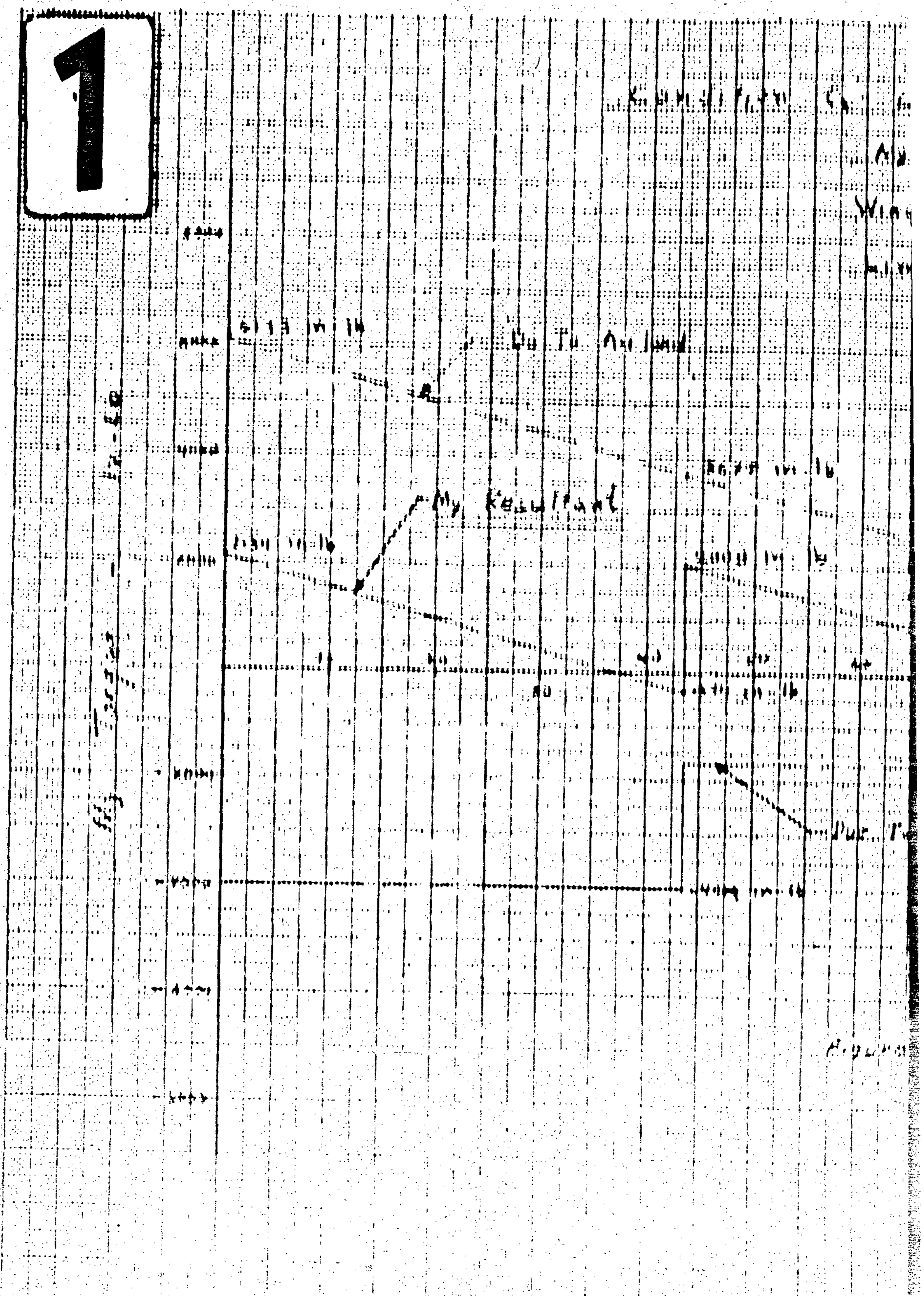

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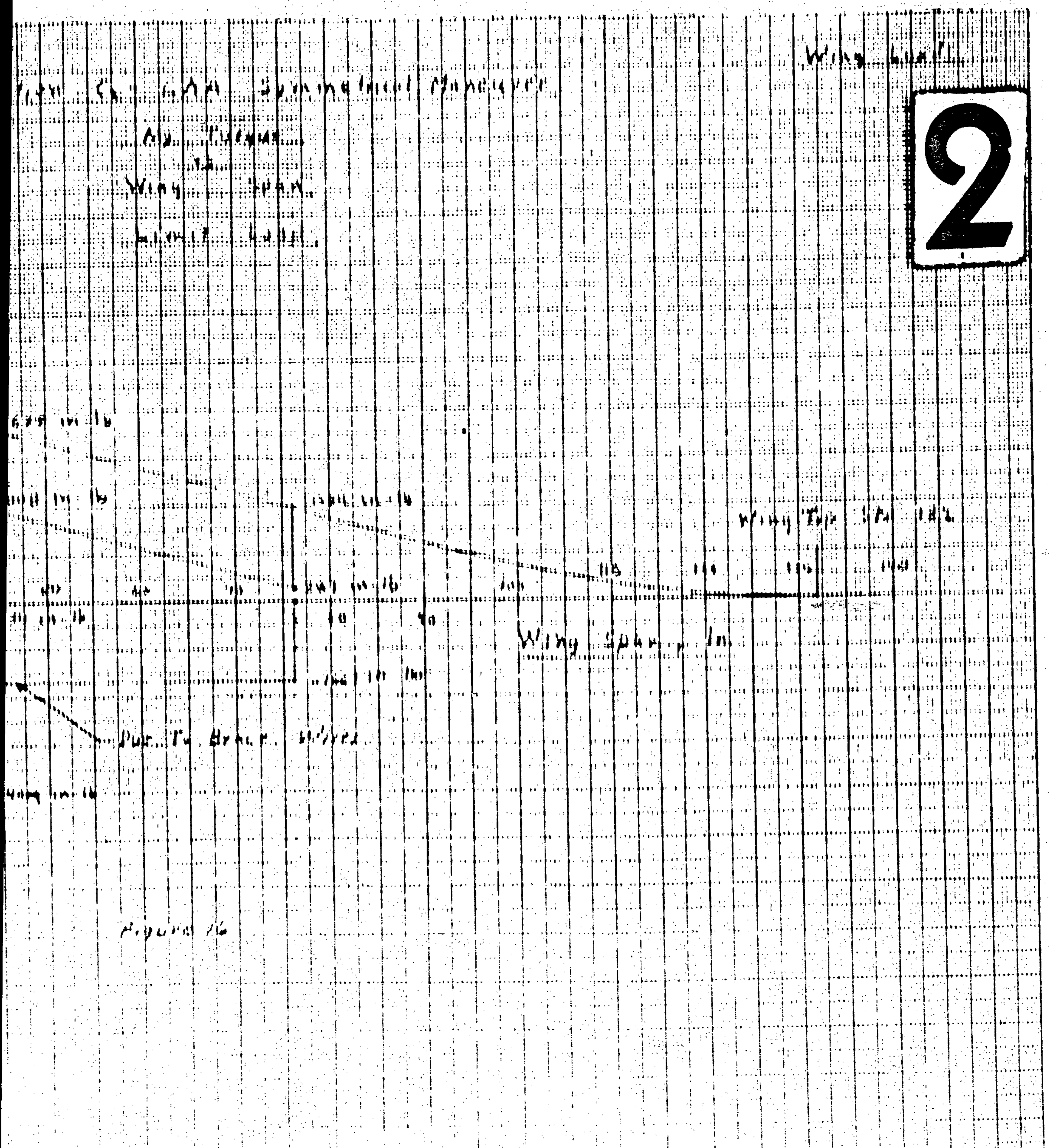

Figure 16

FUSELAGE LOADS

The most important loads are those due to pitching and yawing accelerations and airloads on the horizontal and vertical surfaces. A cable from the engine mount supports the tail when down loads act.

Figure 17 shows the geometry of the fuselage. All the inertia loads are reduced to concentrated forces at points which are numbered 0 to 11. The point 0 is just beneath the cables that fasten the engine mount to the fuselage while point 11 is at the center of the aft spherical end cap. Forces on the horizontal and vertical tail are resolved at point 11 as forces and couples.

The calculations that follow give the rotational inertia loads for pitching about the most forward and aft centroids. The coordinates used for the centroids are early values and do not agree with the final values, however the effect is negligible.

COORDINATES OF CENTROID

		x	z
Early Values	Most Fwd	69.26	47.30
	Most Aft	70.40	47.50
Final Values	Most Fwd	71.76	44.08
	Most Aft	77.58	47.93

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Fuselage Geometry

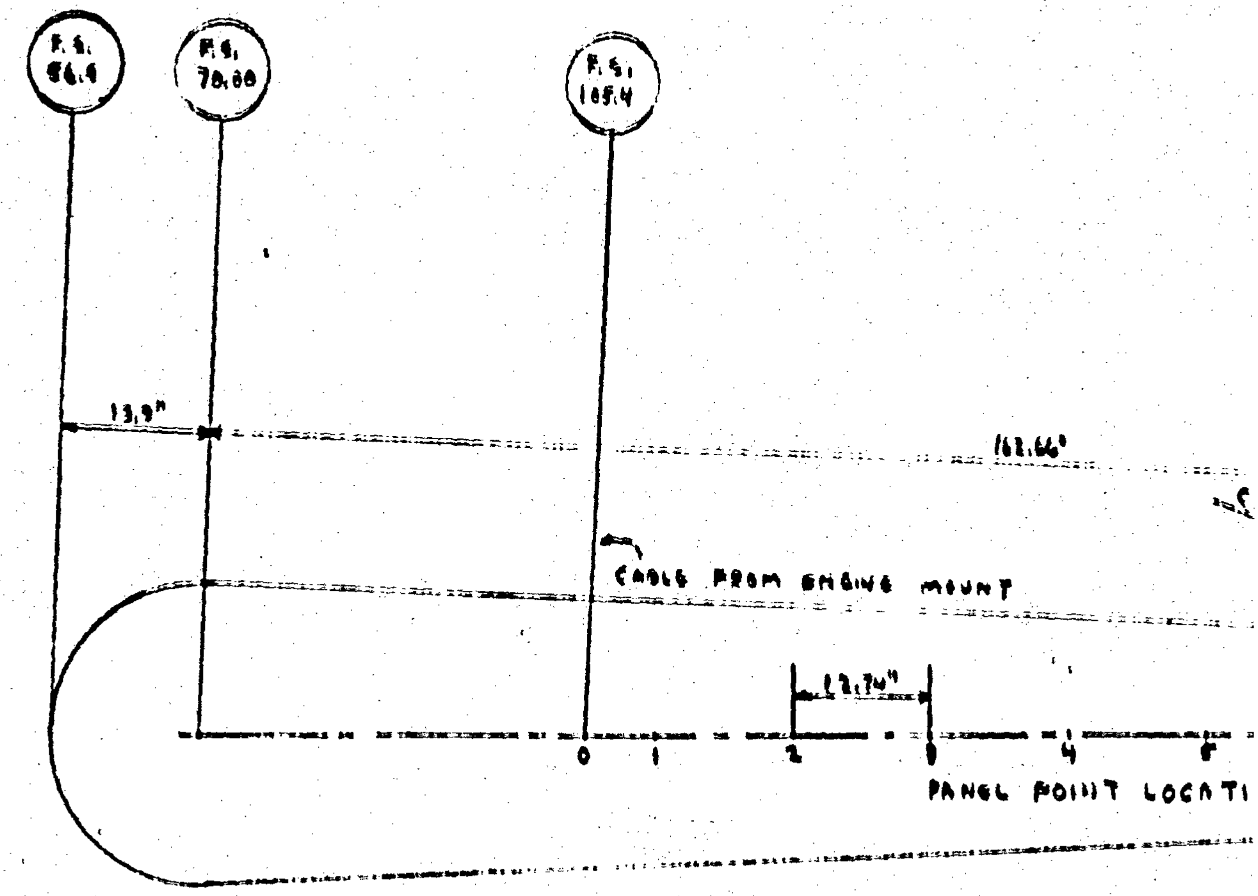

Figure 17

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FUSELAGE LOADS

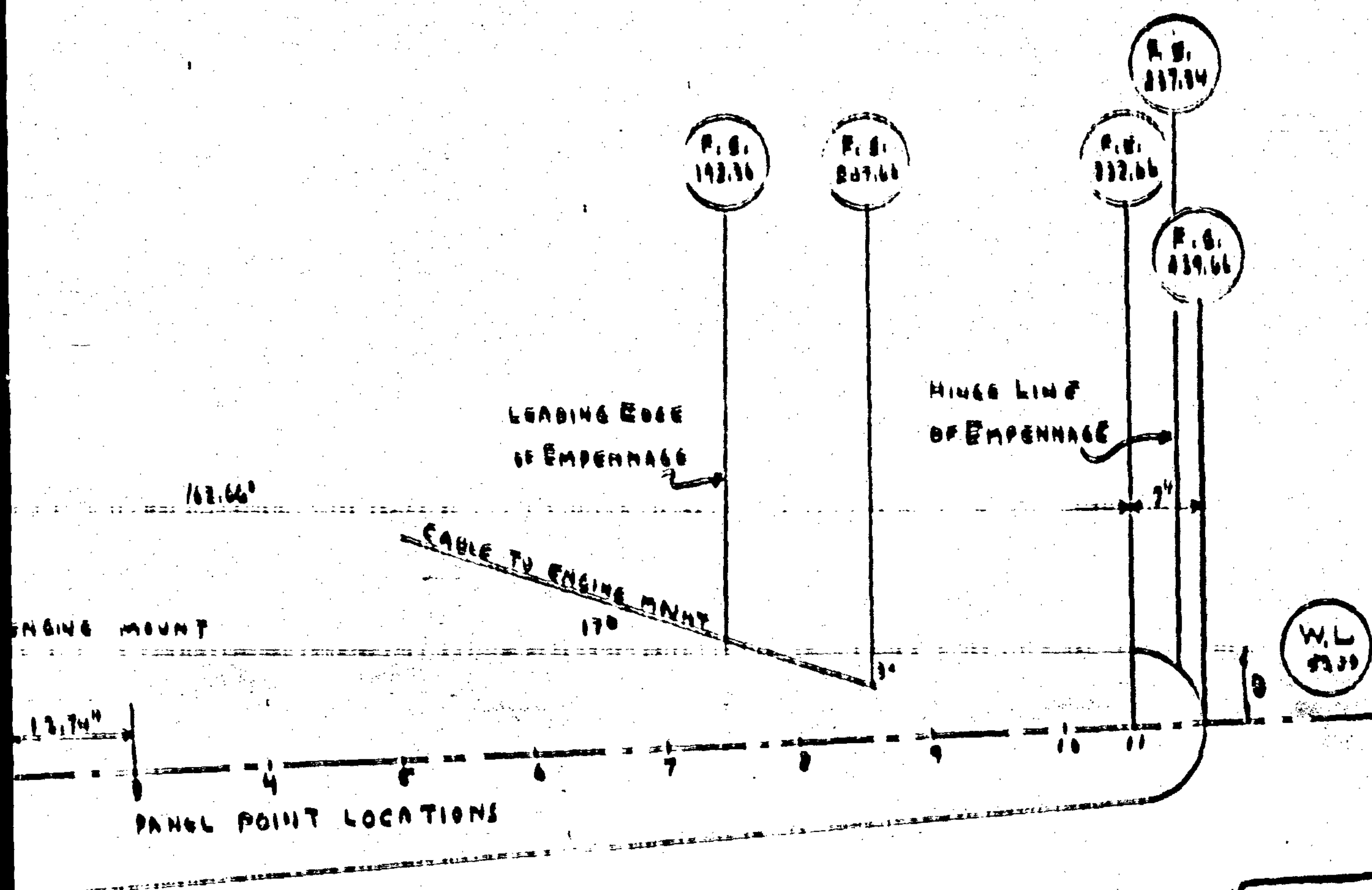

Figure 17

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Rotational inertia loads on fuselage **FUSELAGE LOADS**

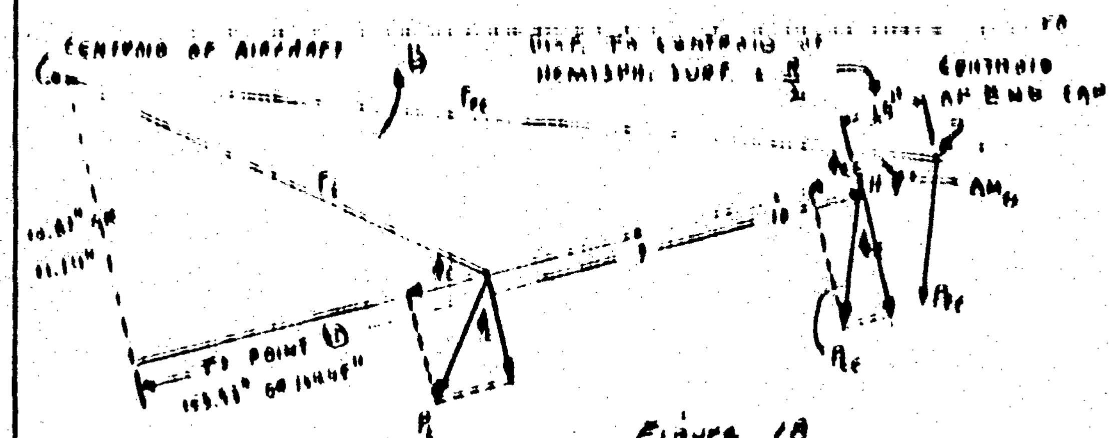

Figure 18

For any panel weight W_i

$$P_i = r_i \ddot{\theta} \frac{W_i}{g}$$

$$P_i \cos\phi_i = \ddot{\theta} \frac{W_i}{g}(r_i \cos\phi_i) \quad \text{BENDING COMPONENT}$$

$$P_i \sin\phi_i = \ddot{\theta} \frac{W_i}{g}(r_i \sin\phi_i) \quad \text{THRUST COMPONENT}$$

For transfer of end cap load, P_{ec}, to point (1)

MOST AFT POSITION OF C.G.	MOST FWD POSITION OF C.G.
$\phi_{ec} = \tan^{-1}\frac{10.87}{153.91+5.5} = 3.96°$	$\phi_{ec} = \tan^{-1}\frac{11.19}{167.96+8.5} = 3.82°$
$r_{ec} = \frac{157.91}{\cos 3.96} = 157.1"$	$r_{ec} = \frac{167.96}{\cos 3.82} = 167.8"$
$P_{ec} = \frac{W_{ec}}{g} r_{ec}\ddot{\theta} = \frac{.061}{32.2}\frac{157.1}{12}\ddot{\theta} = .0248\ddot{\theta}$ LB	$P_{ec} = \frac{.61}{32.2}\frac{167.8}{12}\ddot{\theta} = .265\ddot{\theta}$ LB
Bend Comp = $P_{ec}\cos\phi_{ec}$ $= \ddot{\theta}\frac{W_{ec}}{g}(r_{ec}\cos\phi_{ec}) = .248\ddot{\theta}$ LB	Bend. Comp = $.265\ddot{\theta}$ LB
Thrust Comp = $\frac{\ddot{\theta} W_{ec}}{g}(r_{ec}\sin\phi_{ec}) = .01717\ddot{\theta}$ LB (Compression)	Thrust comp = $.01765\ddot{\theta}$ LB (Compression)
$\Delta M_{ec} = 3.5 P_{ec}\cos\phi_{ec}$ IN-LB $= .868\ddot{\theta}$ IN LB	$\Delta M_{ec} = .928\ddot{\theta}$ IN. LB

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FUSELAGE LOADS

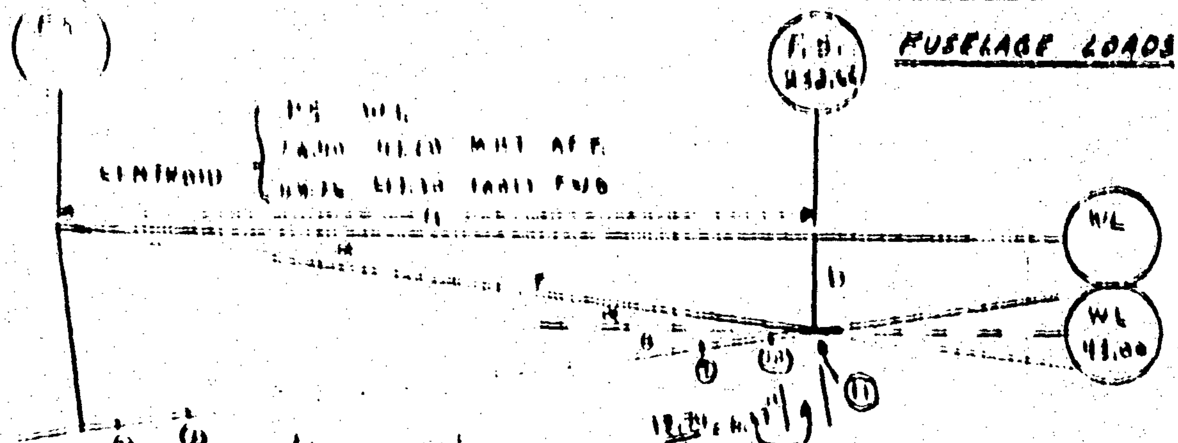

Figure 19

For MAIL AFT POSITION

$a = 233.66 - 78.90 = 154.76''$

$b = 47.30 - 43.00 = 4.30''$

$\alpha = \tan^{-1} \frac{b}{a} = 1.59^\circ$

$\theta = 2.29^\circ$

$\phi = \alpha + \theta = 3.88^\circ$

$r = a \sec \alpha = 154.82$

$r \sin \phi = 10.65''$

$r \cos \phi = 153.93''$

For MAIL FWD POSITION

$a = 233.66 - 68.76 = 164.90''$

$b = 47.30 - 43.00 = 4.30''$

$\alpha = \tan^{-1} \frac{b}{a} = 1.50^\circ$

$\theta = 2.29^\circ$

$\phi = \alpha + \theta = 3.79$

$r = a \sec \alpha = 164.95''$

$r \sin \phi = 10.88''$

$r \cos \phi = 163.96''$

Table XV

F.S.	$r \cos \phi$ MAIL FWD IN.	$r \cos \phi$ MAIL AFT IN.
1	42.93	32.90
2	55.67	45.64
3	68.41	58.38
4	81.15	71.12
5	93.89	83.86
6	106.63	96.60
7	119.37	109.34
8	132.11	122.08
9	144.85	134.82
10	157.59	147.56
11	163.96	153.93

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FUSELAGE LOADS

Transfer of Tail Inertia Load to Point (1)

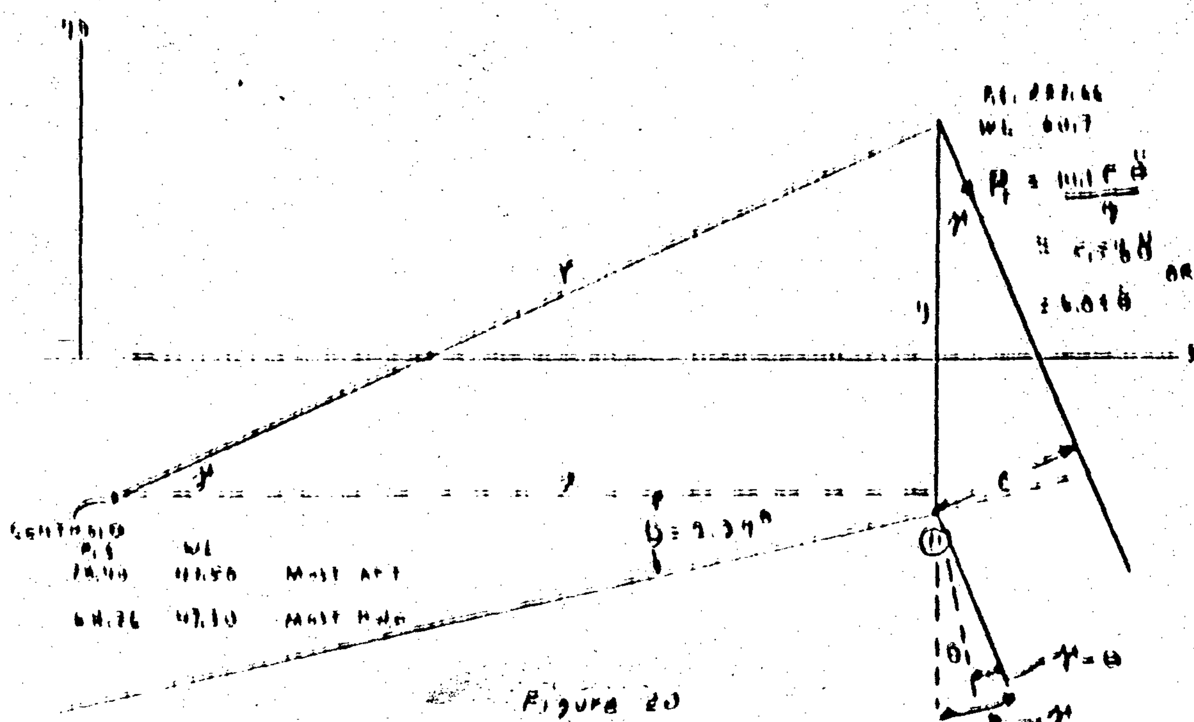


Figure 20

Most Aft Position

$y = 60.70 - 47.53 = 13.10''$

$x = 233.66 - 78.90 = 154.76''$

$\gamma = \tan^{-1} \frac{13.10}{154.76} = 4.91°$

$r = \frac{154.76}{\cos \gamma} = 155.14''$

$c = (60.70 - 47.60) \sin \gamma = 1.52''$

Moment = Pc = 5.54 θ × 1.52 = 8.50 θ in.lb

Bend. Comp = $P \cos(\gamma - \theta)$ = $P \cos 2.64°$

= .9990 P

= 5.54 θ lb

Thrust Comp = $P \sin(\gamma - \theta)$
(Tension)
= .0461 P

= .257 θ lb

Most Fwd Position

$y = 60.70 - 47.30 = 13.40''$

$x = 233.66 - 68.76 = 164.90$

$\gamma = \tan^{-1} \frac{13.40}{164.90} = 4.68°$

$r = \frac{164.90}{\cos \gamma} = 165.4''$

$c = 13.40 \sin \gamma = 1.09''$

Moment = Pc = 6.01 θ × 1.44

= 8.65 θ in.lb

Bend Comp = $P \cos 2.37$

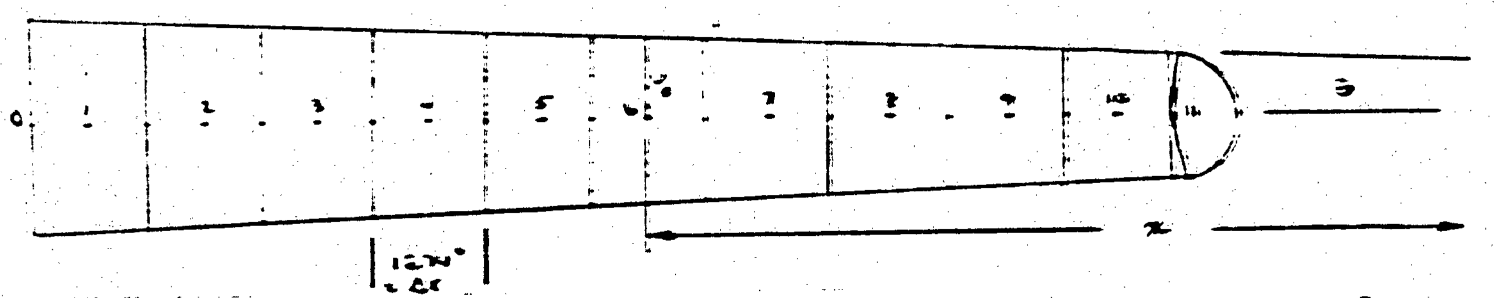
= .9991 P

= 6.01 θ lb

Thrust Comp = .0413 P
(Tension)
= .249 θ lb.

FUSELAGE LOADS

TABLE XVI ROTATIONAL AND TRANSLATIONAL INERTIA LOADS OF FUSELAGE



$y = x \tan\theta$ $y_{10} = 181.50 \tan 2.29° = 7.27"$ Area $= 2\pi y\,\Delta x$ $A_{R} \doteq 2\pi(7)^2 = 303 \text{ in}^2$

		y in	A in²	Wl LB	$r_i \cos\phi_i$ in (For Most Fwd RC)	Bend Comp. LB (For Most Fwd RC)	Thrust Comp LB (For Most Fwd RC)	$r_i \cos\phi_i$ in (For Most Aft RC)	Bend Comp LB (For Most Aft RC)	Thrust Comp LB (For Most Aft RC)	πy^2 in²
	0	12.28									475
	1	12.04	978	1.937	42.93	.215	-.0558	32.9	.165	.0245	457
	2	11.57	935	1.852	55.67	.267	-.0585	45.24	.219	.0272	417
	3	10.98	892	1.765	68.41	.313	-.0556	58.35	.267	.0417	380
	4	10.45	849	1.682	71.15	.353	-.0485	71.12	.309	.0473	344
	5	9.93	806	1.594	93.89	.388	-.0460	83.86	.346	.0447	310
	6	9.40	763	1.510	106.63	.417	-.0435	96.60	.377	.0405	278
	7	8.87	720	1.425	119.37	.440	-.0411	109.34	.403	.0426	248
	8	8.34	677	1.340	132.11	.458	-.0386	122.08	.403	.0377	217
	9	7.80	634	1.255	144.85	.470	-.0362	134.82	.432	.0359	191.5
	10	7.27	591	1.170	157.59	.477	-.0322	147.58	.443	.0335	166.3
END CAP	R_c		302	.610	163.96	.265	-.0176	153.93	.248	.0173	
TAIL	P_T					2.53	+.249		5.78	.297	
END CAP	ΔM					.926			.368		
TAIL	ΔM					3.65			5.53		

$\Sigma = -.2$

NET THRUST AT POINT C IS 0.2# COMP.

NET THRUST BETWEEN POINTS 7 AND 8 IS 0.1# TENSION

THESE THRUSTS ARE NEGLIGIBLE

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FUSELAGE LOADS

Horizontal and Vertical Tail

Horizontal Tail Loads

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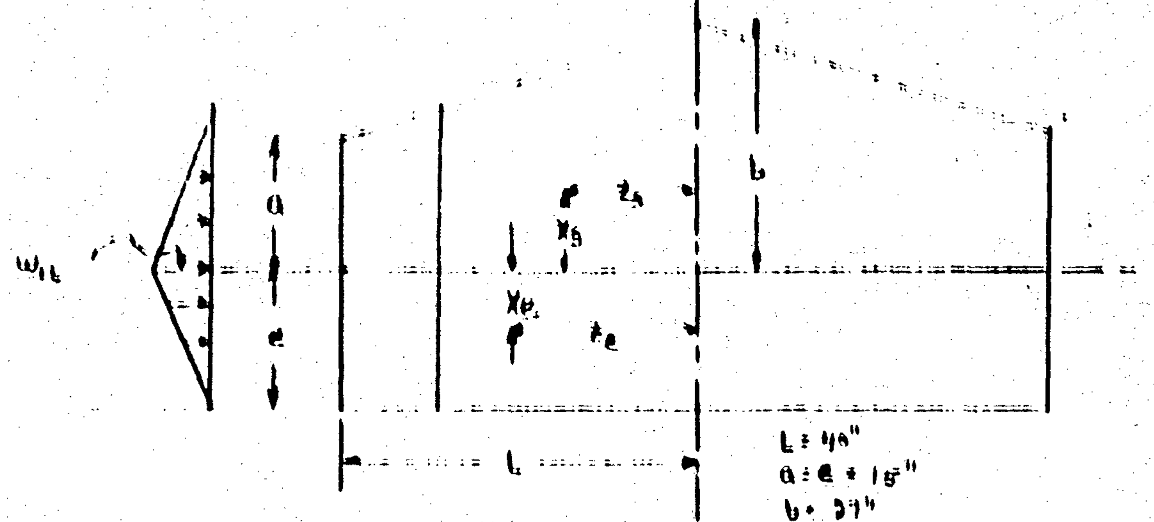

Figure 21

$P_s = \frac{a+b}{4} w_{tl} L$ = Load on stabilizer half span

$P_e = \frac{a}{2} w_{tl} L$ = " " elevator " "

For Whole Tail

$$x_s = \frac{a}{3}\,\frac{b^2+ab+a^2}{b+a} \qquad x_e = \frac{e}{3} \qquad w_{tl} = \frac{4W}{L(a+b+2e)}$$

$$z_s = \frac{2a+b}{3(a+b)} L \qquad z_e = \frac{L}{2}$$

W = Load on half span.

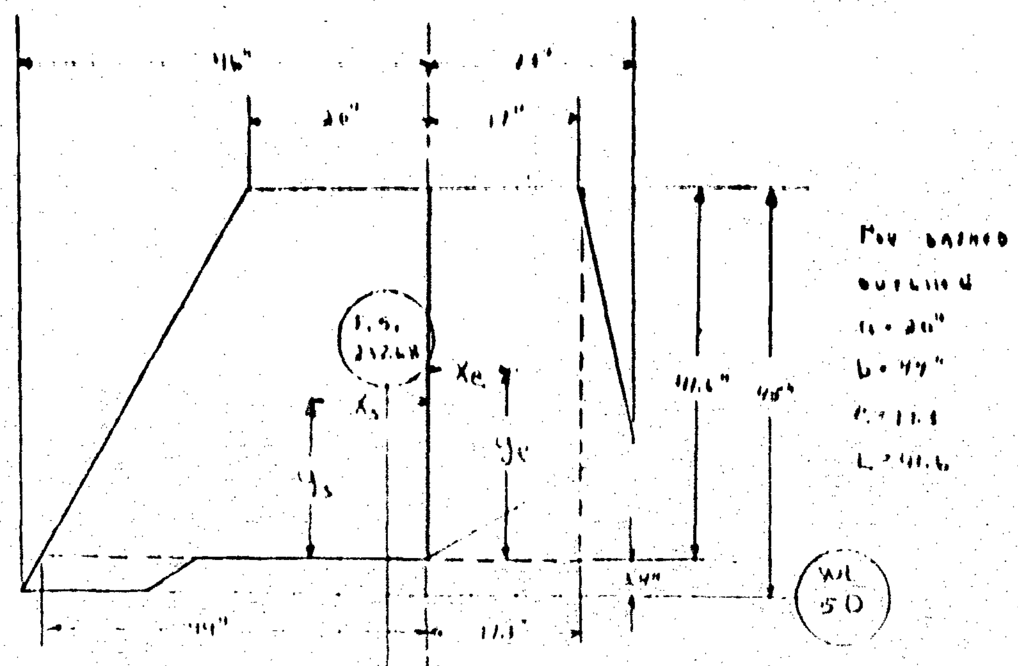

Figure 22

For 100# load on horizontal or vertical tail surfaces

	P_s	Y_s	z_s hinge to C.P.	P_e	X_e	z_e hinge to C.P.
Horizontal Tail	29.1#	1.13"	18.07"	70.9#	8"	7.8"
Vertical Tail	64.8#	11.16"	18.73"	35.2#	8.77"	10.8"

W = 50 lb for horizontal tail
= 100 lb " vertical "

Torque Caused by Unequal Loads on Elevator

Let P_{sL} = load on left stabilizer, P_{sR} = " right " $\Big\}\ P_{sL} + P_{sR} = 2P_s$

From measurements, $\dfrac{P_{sL}}{P_{sR}} = \dfrac{1.2}{.8} = \dfrac{3}{2}$

$T_s = (P_{sL} - P_{sR})\, z_s$ = torque from stabilizer loads
$T_e = (P_{eL} - P_{eR})\, z_e$ = " " elevator "

Then $T_s = \left[\frac{3}{5}(2P_s) - \frac{2}{5}(2P_s)\right] z_s = \frac{2}{5} P_s z_s$

$T_e = \frac{2}{5} P_e z_e$

Net Torque $= T_s + T_e = \frac{2}{5}\left(P_s z_s + P_e z_e\right)$

$= \frac{2}{5}(29.1 \times 18.07 + 70.8 \times 10) = 318$ in-lb

of torque for 100# on the horizontal tail

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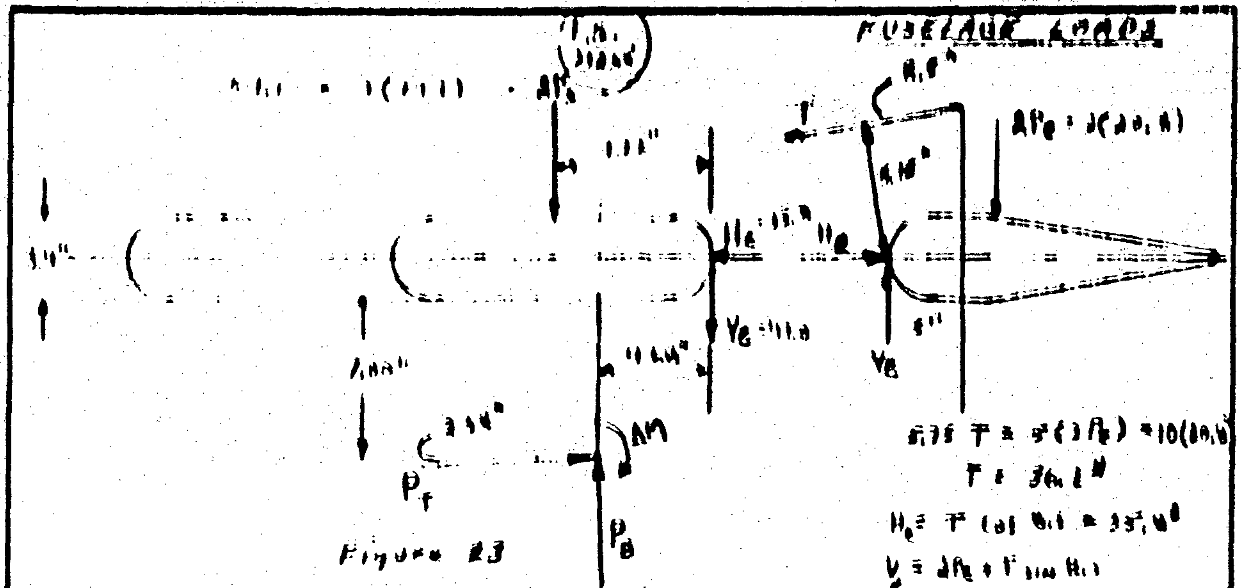
FUSELAGE LOADS



$5.75\,T = 5(2P_t) = 10(20.6)$

$T = 36.1^{\#}$

$H_e = T\cos 8^\circ = 35.8^{\#}$

$V_e = 2P_t + T\sin 8^\circ = 47.0^{\#}$

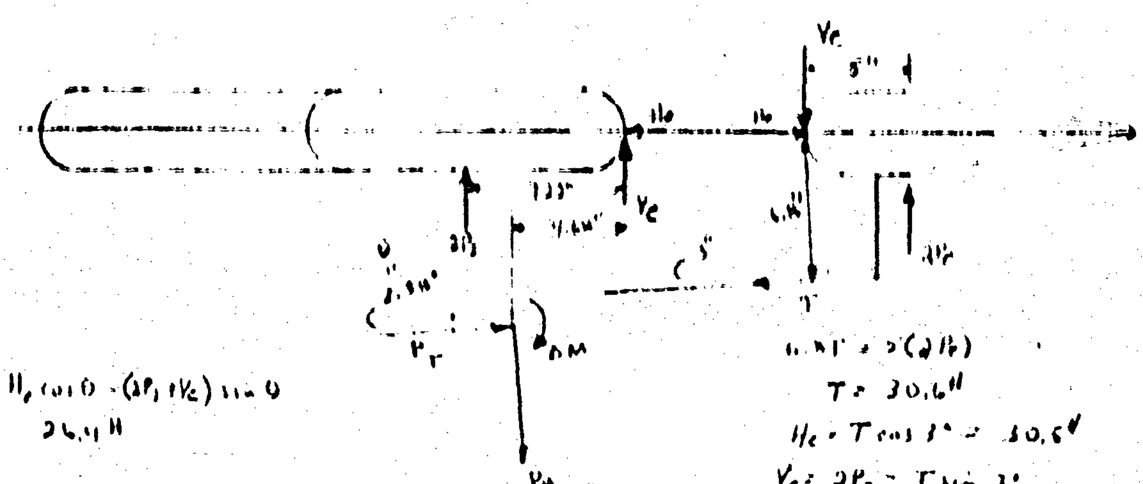
Figure 23

$P_F = (2P_t + V_e)\sin 2.38 + H_e\cos 2.38 = 40.1^{\#}$

$P_B = (2P_t + V_e)\cos 2.38 - H_e\sin 2.38 = 103.9^{\#}$

$\Delta M = (7.22 - 4.68)(2P_t) + (7.88 + 1.1)H_e - 4.68\,V_e = 239 \text{ in.}\#$

CALCULATIONS FOR $100^{\#}$ DOWN



$P_T = H_e\cos\theta - (2P_t + V_e)\sin\theta = 26.4^{\#}$

$6.87\,T = 5(2P_t)$

$T = 30.6^{\#}$

$H_e = T\cos 3^\circ = 30.5^{\#}$

$V_e = 2P_t - T\sin 3^\circ = 110^{\#}$

$P_B = (2P_t + V_e)\cos\theta + H_e\sin\theta = 100.5^{\#}$

Figure 24

$\Delta M = 2P_t(7.22 - 4.68) + (11.1)H_e \;\; 14.68\,V_e = 302 \text{ in-lb}$

REACTIONS AT (FS 337.8, WL 43.00) DUE TO UP AND DOWN LOADS IN HORIZONTAL TAIL. LOADS ARE FROM PREVIOUS PAGE.

CALCULATIONS FOR $100^{\#}$ UP

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CALCULATIONS FOR 100# ON VERTICAL TAIL

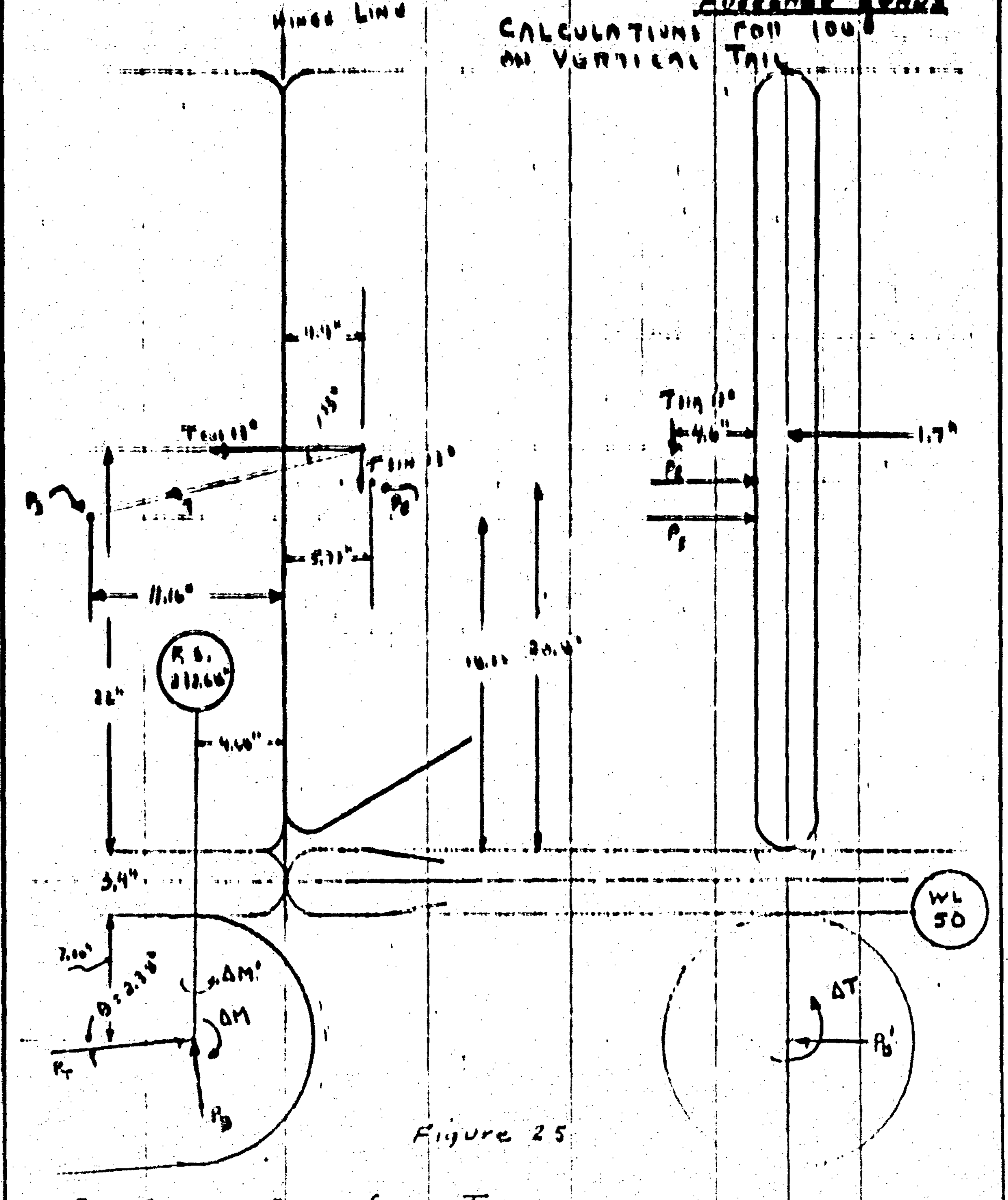

Figure 25

FOR CONTROL FORCE CABLE TENSION

$\Sigma M_{HINGE\ LINE} = 0 \qquad (T \cos 13)(4.4+1.7) = 5.77 P_c$

$T \cos 13° = 37.2^{\#}$

$T \sin 13° = 7.15^{\#}$

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FUSELAGE LOADS

CALCULATIONS FOR 100# ON VERTICAL TAIL

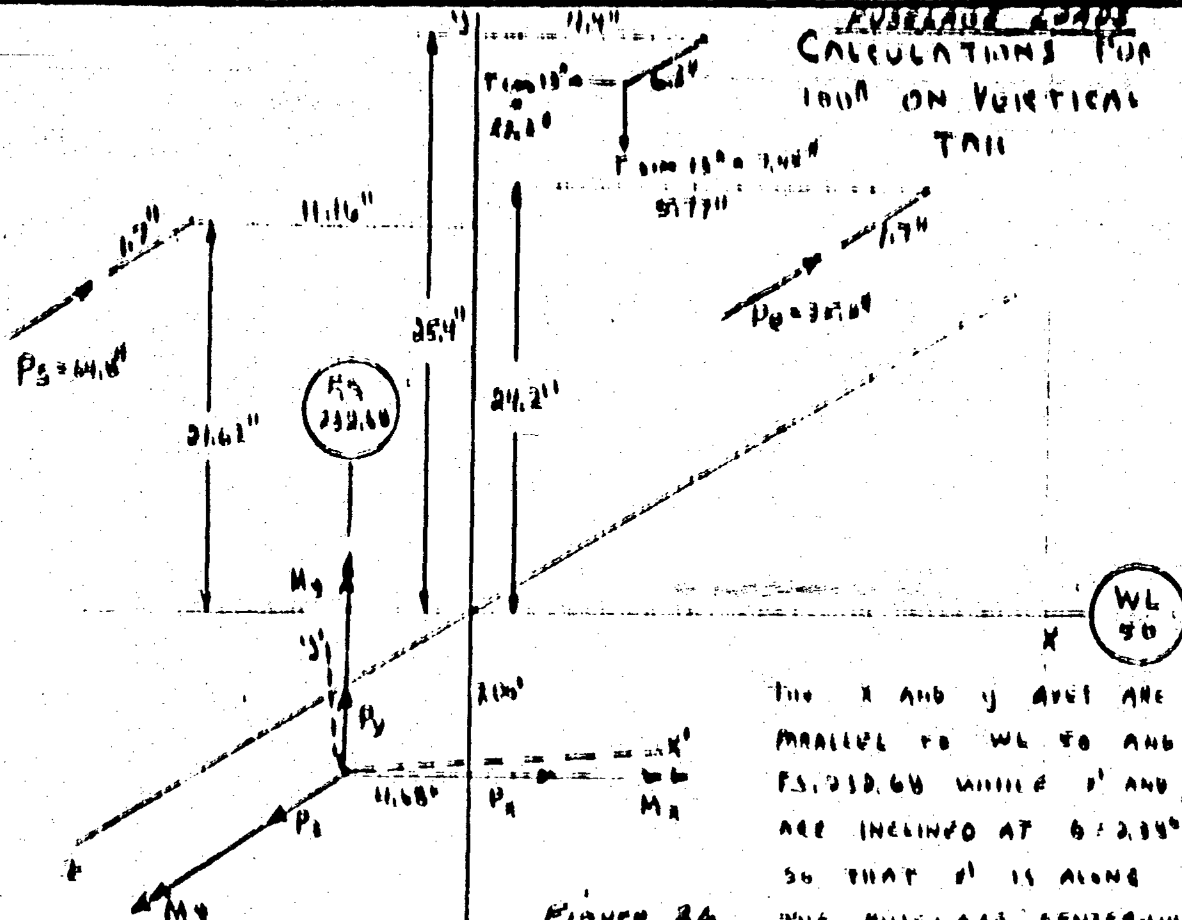

Figure 26

THE X AND Y AXES ARE PARALLEL TO WL 90 AND F.S. 232.68 WHILE x' AND y' ARE INCLINED AT θ = 2.35° SO THAT x' IS ALONG THE FUSELAGE CENTERLINE

REACTIONS AT WL 93.00 F.S. 232.68

$P_x = T \cos 13° = 97.7$#

$M_x = -6.3\, T \sin 13 + 31.21 P_6 + 24.21 P_5 = 2910$ in-#

$P_y = T \sin 13° = 22.5$#

$M_y = 6.3\, T \cos 13 - (9.77 + 11.68) P_6 + (11.16 - 11.68) P_5 = 255$ in-#

$P_z = P_6 + P_5 = 100$# SHEAR

$M_z = -(7.06 + 25.4)(T \cos 13) + (11.68 + 11.4)\, T \sin 13° = -977$ in-# BEND

WITH RESPECT TO THE INCLINED AXES

$P_{x'} = P_x \cos\theta + P_y \sin\theta = 98.4$# CAUSES COMP

$M_{x'} = M_x \cos\theta + M_y \sin\theta = 2911$ in-# " TORSION

$P_{y'} = P_y \cos\theta - P_x \sin\theta = 18.1$# SHEAR

$M_{y'} = M_y \cos\theta - M_x \sin\theta = 137$ in-# BEND

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FUSELAGE LOADS.

VERTICAL SHEAR FORCE CAUSED BY BENDING STRESSES

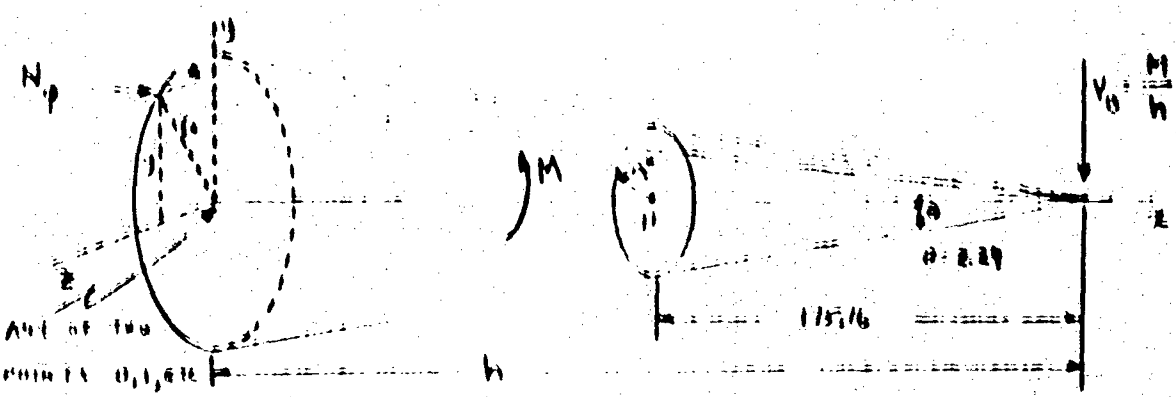

Figure 27

LET N_ϕ = BENDING STRESS, LB./IN., CAUSED BY A MOMENT M. IF THE BENDING STRESSES ARE ASSUMED TO ACT ALONG THE GENERATRIX OF THE CONE, THEN THESE BENDING STRESSES ACT THROUGH A COMMON POINT, THE APEX OF THE CONE. THE STRESSES N_ϕ, WHOSE RESULTANT AT THE Y-Z PLANE IS A FORCE V_0 AND A COUPLE M, CAN BE RESOLVED AT THE APEX OF THE CONE INTO A SINGLE FORCE V_0 WHOSE VALUE MUST BE

$$V_0 = \frac{M}{h}$$

AND WHOSE DIRECTION IS DOWN WHEN THE MOMENT M CAUSES COMPRESSION ON THE UPPER SURFACE.

THE VALUES OF h AND 1/h ARE GIVEN BELOW.

Table XVII

POINT	0	1	2	3	4	5	6	7	8	9	10	11
h, IN.	302.6	296.2	281.5	270.7	258.0	245.2	233.5	219.8	207.0	194.3	181.5	175.7
1/h, IN.	.00331	.00347	.00353	.00369	.00387	.00408	.00430	.00455	.00483	.00514	.00551	.00570

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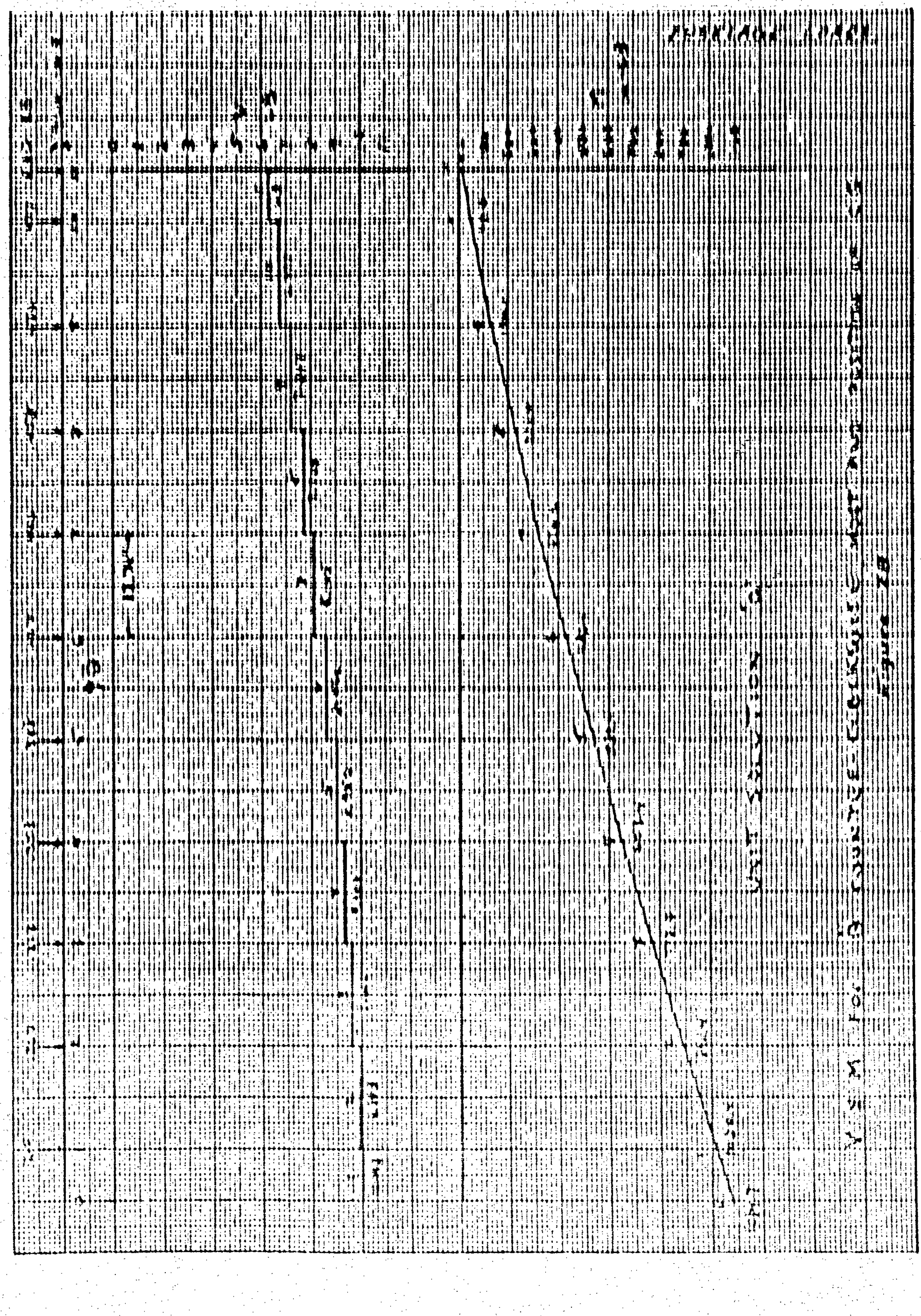

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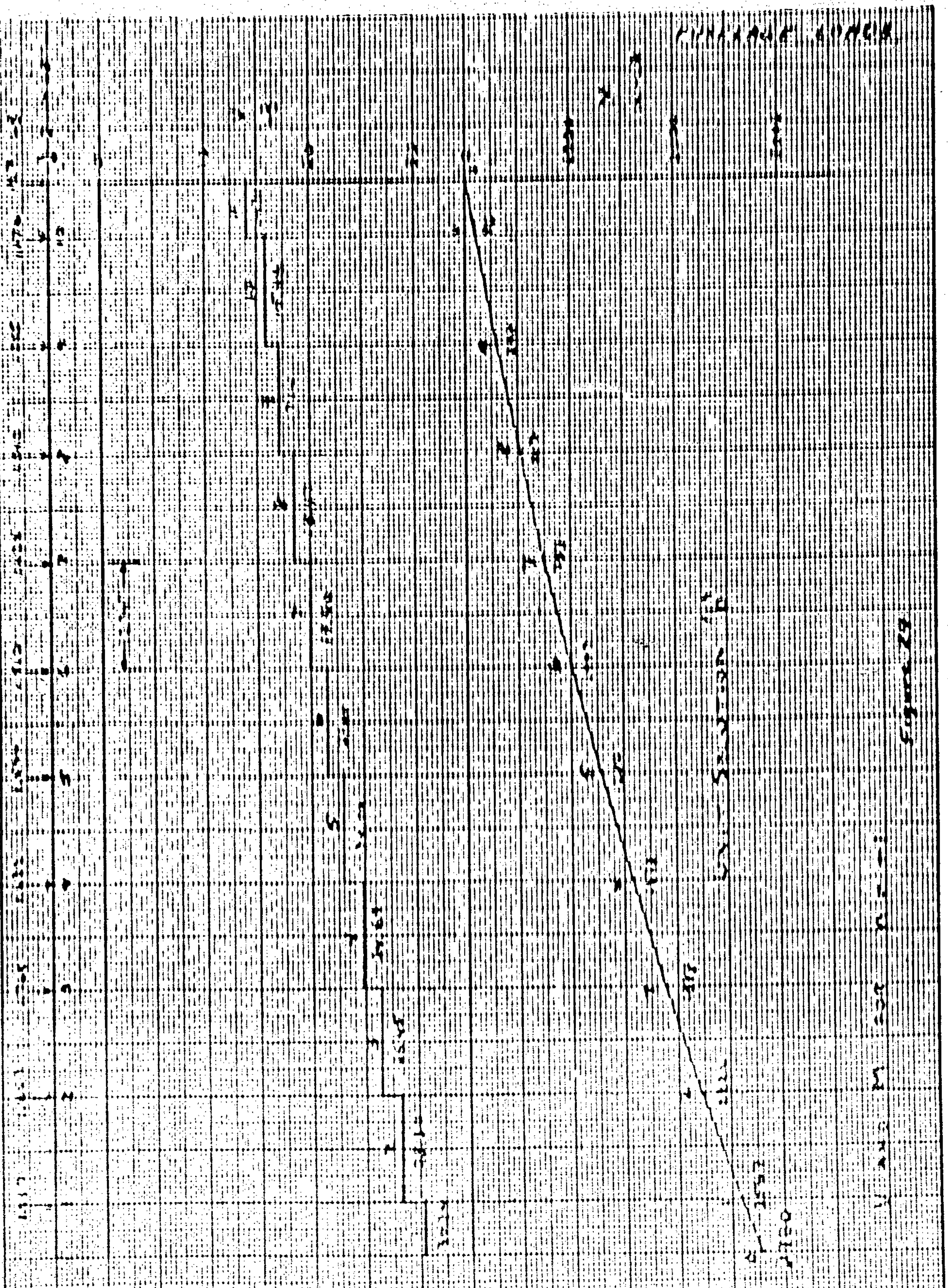

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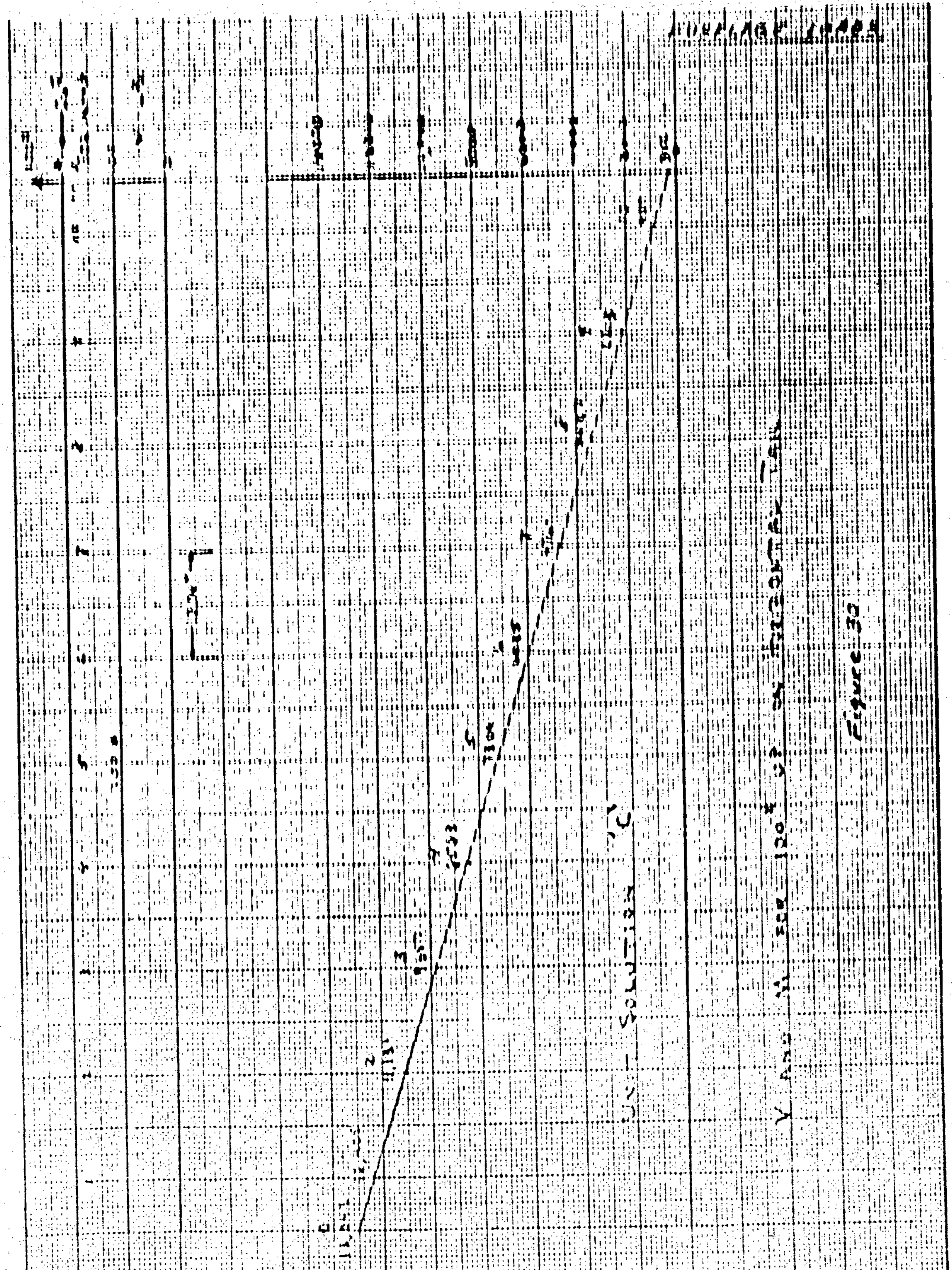

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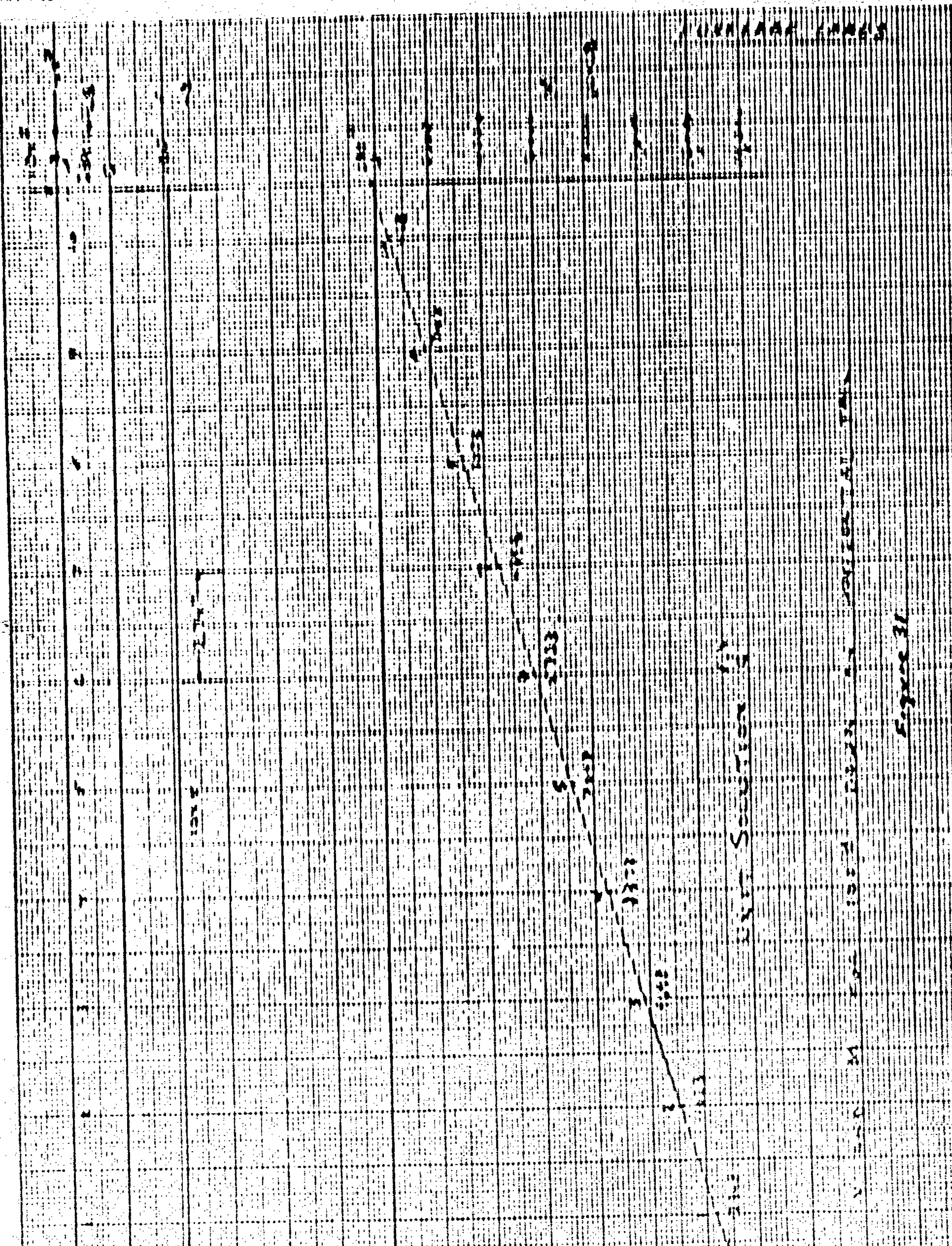

Figure 31

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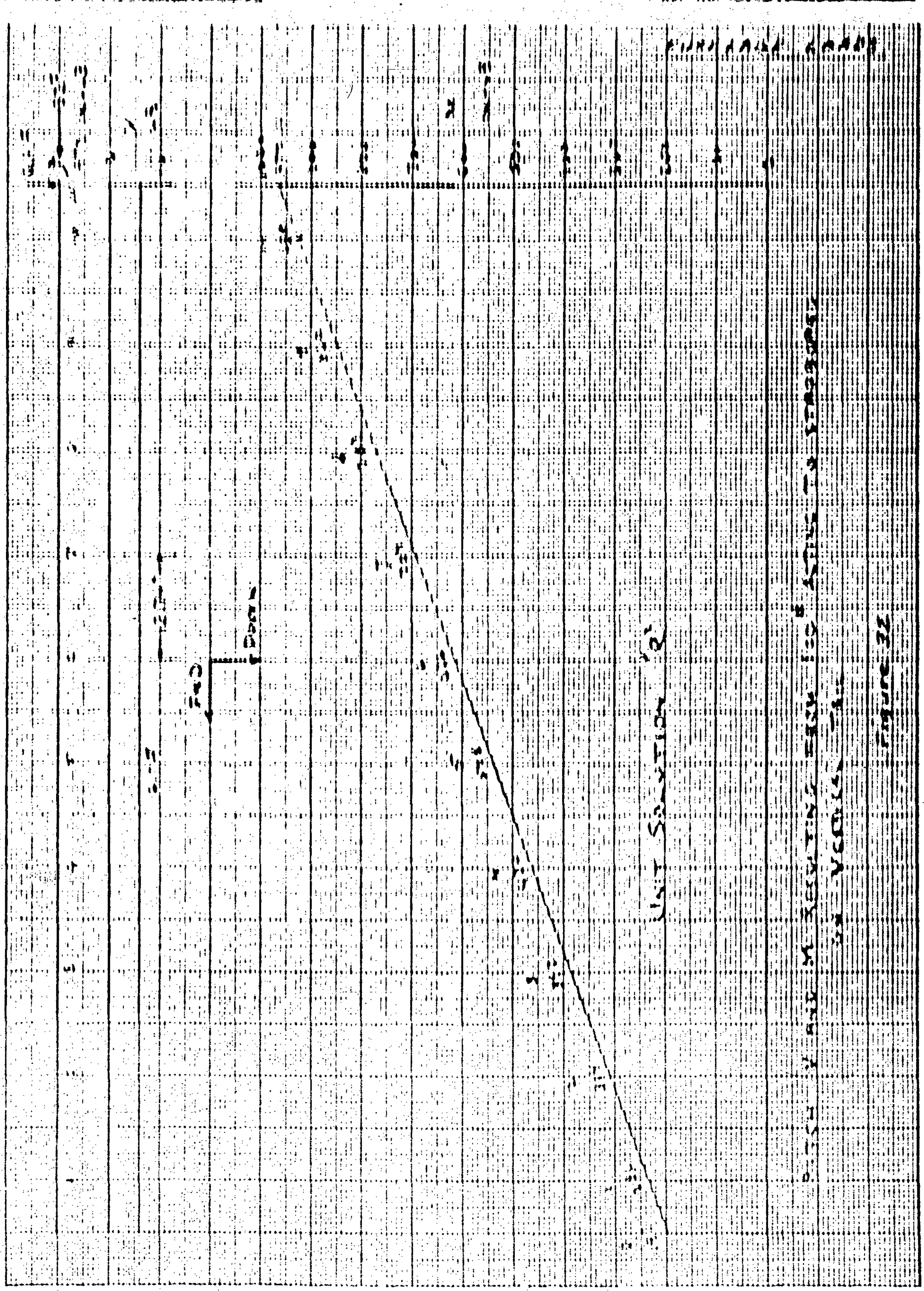

Figure 32

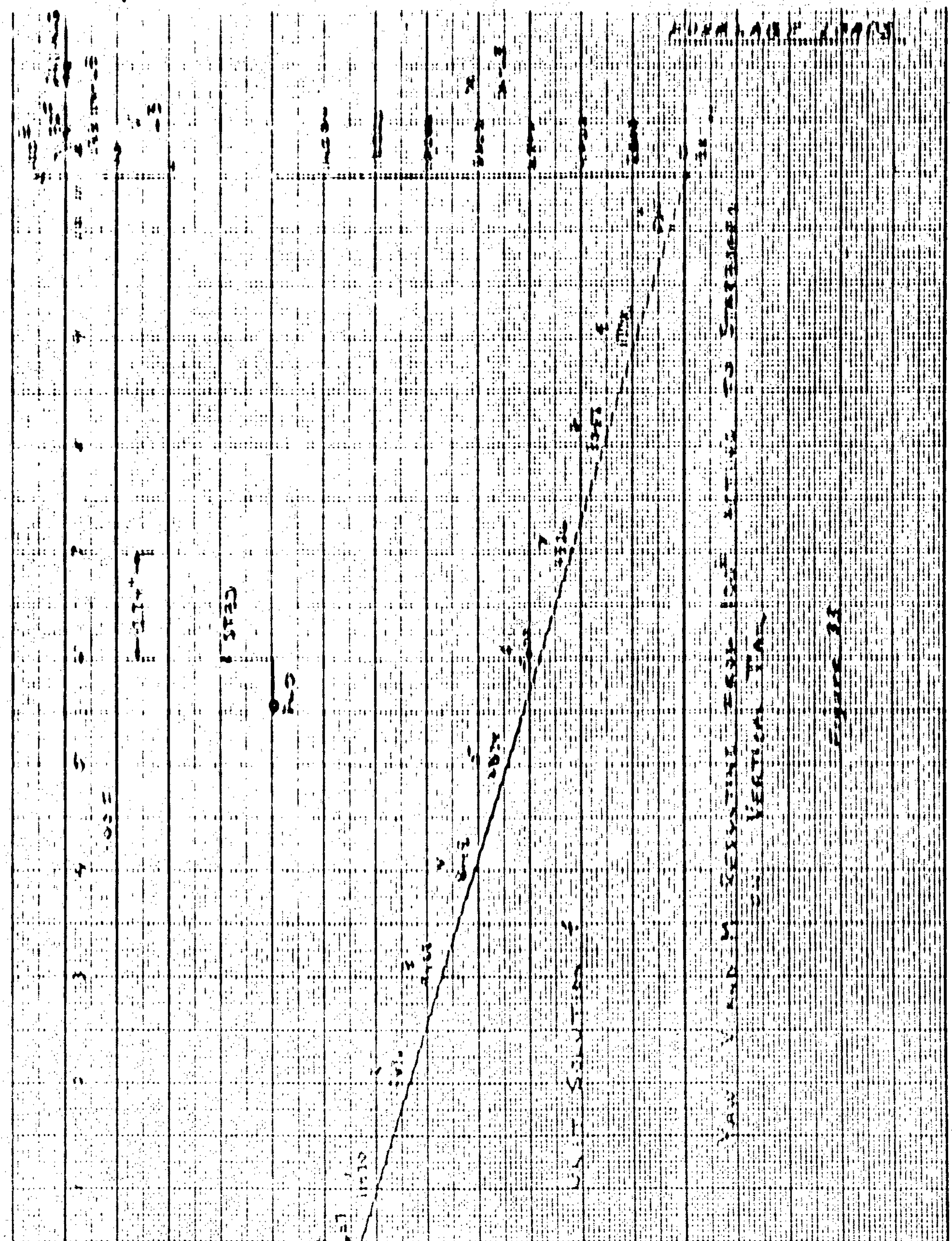

Figure 33

FUSELAGE LOADS

CRITICAL CONDITIONS

From the summary of airloads the most critical conditions are A2, A4, A6, and A7; B6 and B7; C7; E2; F2, F4, F6, F7, and F11 + horizontal tail load of F13. Conditions 3 and 5 are not critical because engine thrust loads up the tail cable.

As the unit solutions are labeled (a) through (f) then these solutions may be combined according to the equations

$$V \text{ or } M = -a\theta + nb + P/100\ c \qquad \text{For conditions 2, 4, 6, and 7}$$

$$V \text{ or } M = -a\theta + nb - P/100\ d \qquad \text{For conditions 3, 5, and 7}$$

The combined condition is given by the vector sum of pitching and yawing shears and moments and the superposition of the torque.

$$\left[V \text{ or } M\right]_{pitch} = nb - \frac{P}{100} d + \frac{P_{yaw}}{100} e$$

$$\left[V \text{ or } M\right]_{yaw} = -a\theta + \frac{P_{yaw}}{100} f$$

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FUSELAGE LOADS

Table XVIII

Reference Point	Values from Unit Solutions a	b	c or d	Loading Quantities $\ddot{\theta}$	Y_1	P	V (LB)	M (IN-LB)	Condition	
u	-6.3	-147	106	-1.83	+2.5	+60	+11.9		A2	
o	-10.1	-30.2	100				-34			
u	-9.1	-2	301					+159		
o	-1085	-2760	+1354					-653		
u	"	"	"	0	2	60	30.6		A9	
o							-14			
u								177		
o								2365		
u	"	"	"	0	3.5	71	+34.2		A6	
o							-4.2			
u								+269		
o								+2300		
u	"	"	"	0	2	71	+40.6		A7	Plus Torque
o							+11.4			
u								+210		
o								-3690		
u	"	"	"	0	2.5	80	42.9		8C	
o							4.1			
u								236		
o								3990		
u	"	"	"	0	2	80	53.6		87	Plus Torque
o							19.6			
u								238		
o								[illegible]		

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TABLE XII FUSELAGE LOADS

Reference Point	Values from Unit Solutions a	b	c	Loading Quantities [illegible]	[illegible]	P	V (lb)	M (in-lb)	Condition	
[illegible]	-[illegible]	-14.7	+[illegible]	[illegible]	[illegible]	70	+[illegible]			
	-[illegible]	-30.2	+100				+9[illegible]		C7	
	-4.6	-[illegible]	+357					+237		
	-[illegible]	-2750	+[illegible]					-[illegible]		
[illegible]	.	.	.	-2	-1	114	-[illegible]		F2	-3F
							+124.2	+227		[illegible]
								-15492		
[illegible]	.	.	.	-2	0	114	+[illegible]		F4	-[illegible]
							+52.2	+325		[illegible]
								-[illegible]		
[illegible]	.	.	.	0	-1	70	+347		F6	
							+[illegible]	-243		
								-[illegible]		
[illegible]	.	.	-104			-114	-[illegible]		F5	+[illegible]
			-104				-43.5	+292		[illegible]
			+239					-[illegible]		
			-13010							
[illegible]	.	.	.	0	0	-70	-72.8		F7	-[illegible]
							-72.5	+[illegible]		[illegible]
								-9128		

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FUSELAGE SPARS

COMBINED CONDITION FII, USING CONDITION BII FOR VERTICAL TAIL LOAD

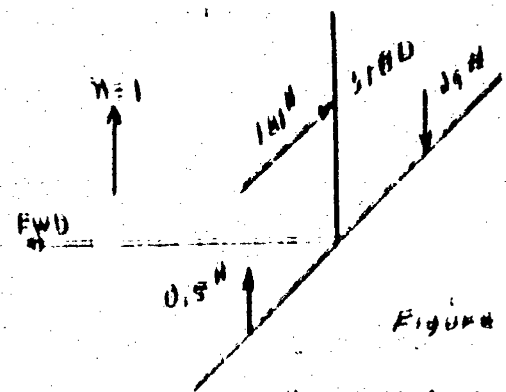


Figure 34

$$\text{Pitch } VEM = nb - \frac{P_{UP}}{100} d + \frac{P_{SIDE}}{100} e$$

$$= b - \frac{(21-.5)}{100} d + 1.81e$$

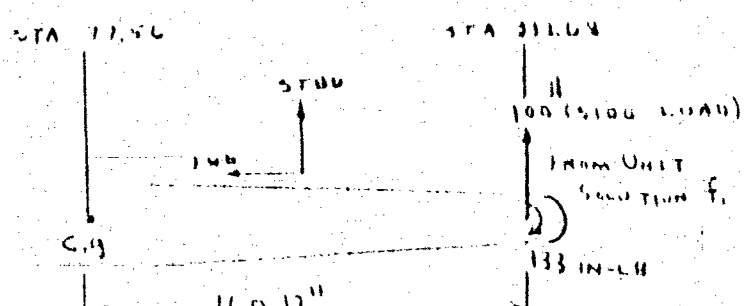
$$= b - .205d + 1.81e$$

$$\text{Yaw } VEM = -a\ddot{\gamma} + \frac{P_{SIDE}}{100} f$$

$$= -a\ddot{\gamma} + 1.81f$$

ACTUALLY THE VALUES 'a' ARE FOR $\ddot{\theta}$ AND NOT $\ddot{\gamma}$ BUT THE ERROR MADE IN USING 'a' IS INSIGNIFICANT

$\ddot{\gamma}$ MAY BE CALCULATED FROM THE YAWING MOMENT OF THE 181# SIDE LOAD AND THE YAWING MOMENT OF INERTIA BOTH ABOUT AN AXIS OF YAW THROUGH THE CENTROID OF THE AIRCRAFT. THE PRINCIPAL AXES OF PITCH AND YAW ARE INCLINED AT ABOUT 20° WITH THE GEOMETRIC AXES SO THAT A YAWING MOMENT ABOUT THE GEOMETRICAL AXIS ACTUALLY CAUSES THE AIRCRAFT TO PITCH AND YAW BOTH ABOUT THE PRINCIPAL AXES.



$$M_{YAW} = \left(100\,\frac{160.12}{12} - \frac{133}{12}\right)\frac{181}{100}$$

$$= 2390 \text{ FT-LB}$$

$$\ddot{\gamma} = \frac{M_{YAW}}{I_{YAW}} = \frac{2390}{300.8} = 8 \text{ rad/sec}^2$$

Figure 35

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FUSELAGE LOADS

PITCH V AND M

1	2	3	4	5	6	7	8	9	10	11	← CORRESPONDS TO VALUE ON CHART
[illegible]	-24.3	[illegible]	[illegible]	-23.0	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	= wb = b
-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	-29.7	= d = -29.7
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	= e = [illegible]
-71.0	-69.1	-67.0	-65.8	-63.3	-62.2	-60.7	-59.3	-57.9	-55.7	-53.5	= V lb

0	1	2	3	4	5	6	7	8	9	10	11	
-3730	-2940	-2255	-1790	-1570	-1390	[illegible]	-723	-522	-340	-91	-2	= wb = b
-3740	-3550	-3140	-2760	-2350	-2015	-1625	-1257	-891	-533	-174	-58	= -.2477 d
+365	[illegible]	+875	+1110	+1252	[illegible]	[illegible]	+1315	[illegible]	+1560	+1520	+1770	= 1.842
-4160	-5680	-4700	-3940	-3110	-2295	-1510	-740	-24	+760	+1517	+1702	= M_P in-lb

YAW V AND M

1	2	3	4	5	6	7	8	9	10	11	
-80.6	-78.0	-76.8	-74.3	-71.5	-68.7	-65.1	-61.5	-58.3	-54.2	-50.1	= $-a\ddot{Y}$ = -36
+181	181	181	181	181	181	181	181	181	181	181	= 1.31 f
1005	1032	1042	1067	1095	1123	1152	1192	1233	1268	1306	= V

0	1	2	3	4	5	6	7	8	9	10	11	
-5520	-8150	-7150	-6130	-5190	-4260	-3420	-2570	-1810	-1130	-396	-76	= $-Q\ddot{T}$ = -8Q
22800	21000	19330	17630	14730	12930	10450	7920	5520	3270	912	-243	= 1.131 f
14160	13500	12230	10,920	9540	7150	6710	5150	3720	2140	916	-316	= M_Y in-lb
15400	14420	13130	11570	10050	[illegible]	6980	5140	3710	2270	1002	1232	= $M = \sqrt{M_P^2 + M_Y^2}$ in-lb

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FUSELAGE LOADS

CORRECTION OF BEAM SHEAR FOR TAPER - PITCH

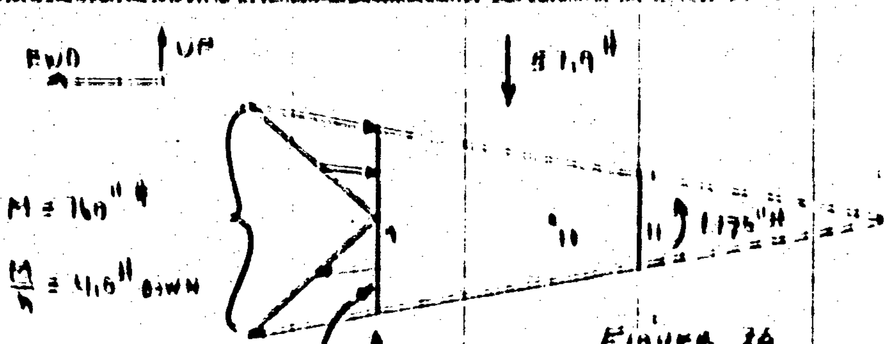

FIGURE 36

SECTION JUST FWD OF POINT 9

$V_g = 57.9 + \frac{760}{168} = 57.4 + 4.5 = 61.9^{\#}$ UP

THUS $V_g = V_{DIAG} - \frac{M}{h} = -57.9 - 4.5 = 61.9^{\#}$ UP

FOR A SECTION JUST FWD OF POINT 6

$V_g = -62.2 - \frac{-1510}{226} = -62.2 + 6.7 = -55.4^{\#}$ UP.

	0	1	2	3	4	5	6	7	8	9	10	11	
	-76.0	-76.0	-69.1	-67.1	-65.9	-63.8	-61.3	-60.4	-59.3	-57.9	-56.7	-55.6	V
	30.8	19.6	17.3	14.9	12.4	9.6	6.7	3.9	-.1	-4.0	-8.9	-10.1	$-\frac{M}{h}$
	-50.3	-51.4	-51.8	-52.2	-53.1	-54.2	-55.5	-57.2	-59.3	-61.9	-60.2	-65.6	V_g

CORRECTION OF BEAM SHEAR FOR TAPER - YAW

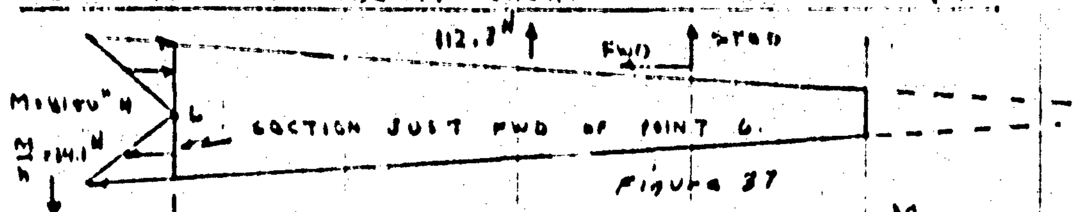

FIGURE 37

$V_g = 112.3 - \frac{6880}{226} = 81.8^{\#} = V_{DIAG} - \frac{M}{h}$

0	1	2	3	4	5	6	7	8	9	10	11	
100.4	100.4	102.0	104.2	106.7	109.5	112.3	115.9	119.5	123	126.8	130.6	V
-48.0	-46.7	-44.2	-41.3	-38.0	-34.1	-30.5	-24.1	-18.6	-11.4	-2.9	+1.9	$-\frac{M}{h}$
52.4	53.7	57.8	62.9	68.7	75.4	81.8	91.8	100.9	111.6	123.9	132.5	V_g

$V = \sqrt{(V_g)^2_{PITCH} + (V_g)^2_{YAW}}$

0	1	2	3	4	5	6	7	8	9	10	11
72.4	74.5	77.1	81.3	86.3	93.0	99.0	108.2	117.0	127.5	140.0	140.0

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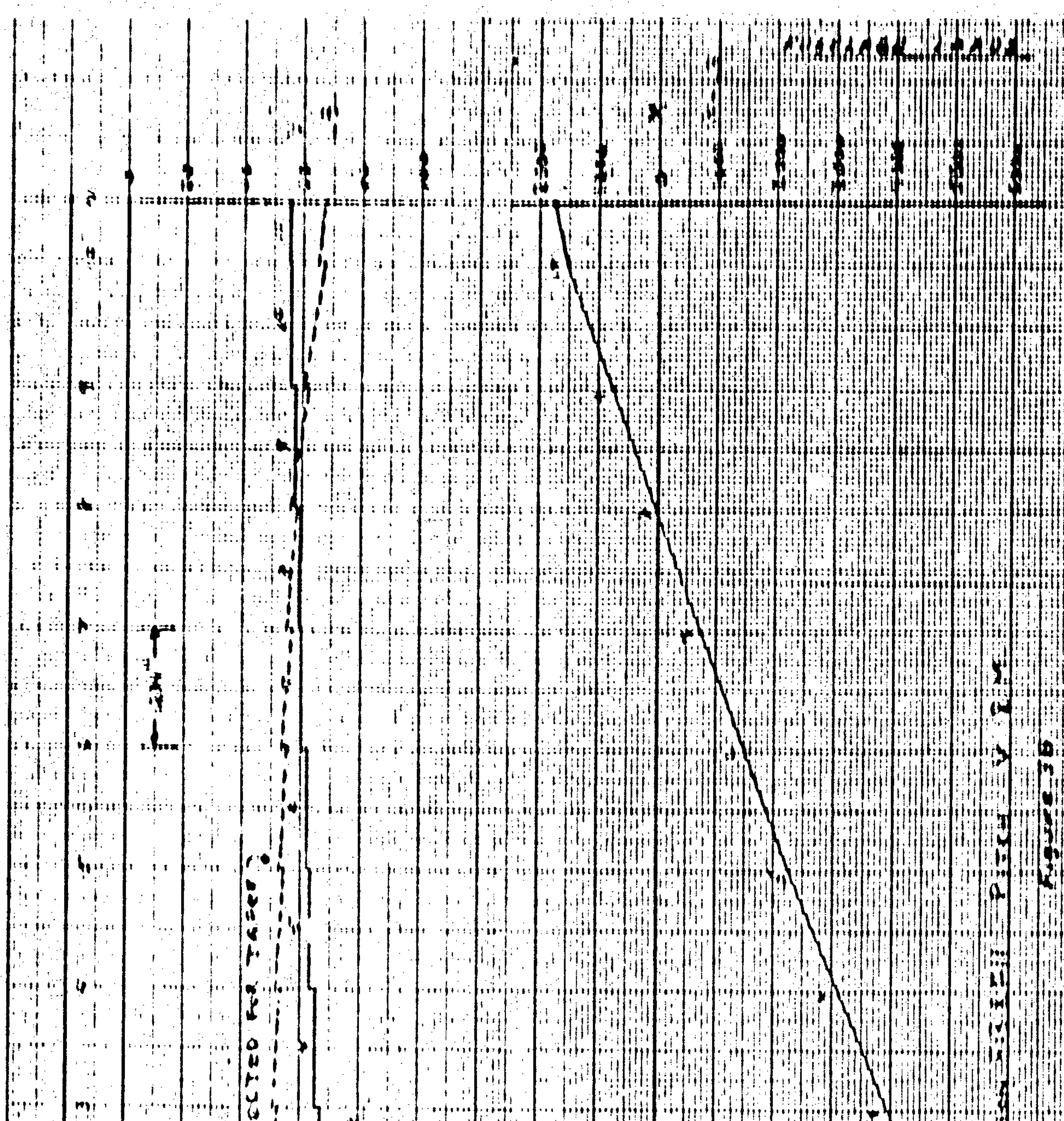

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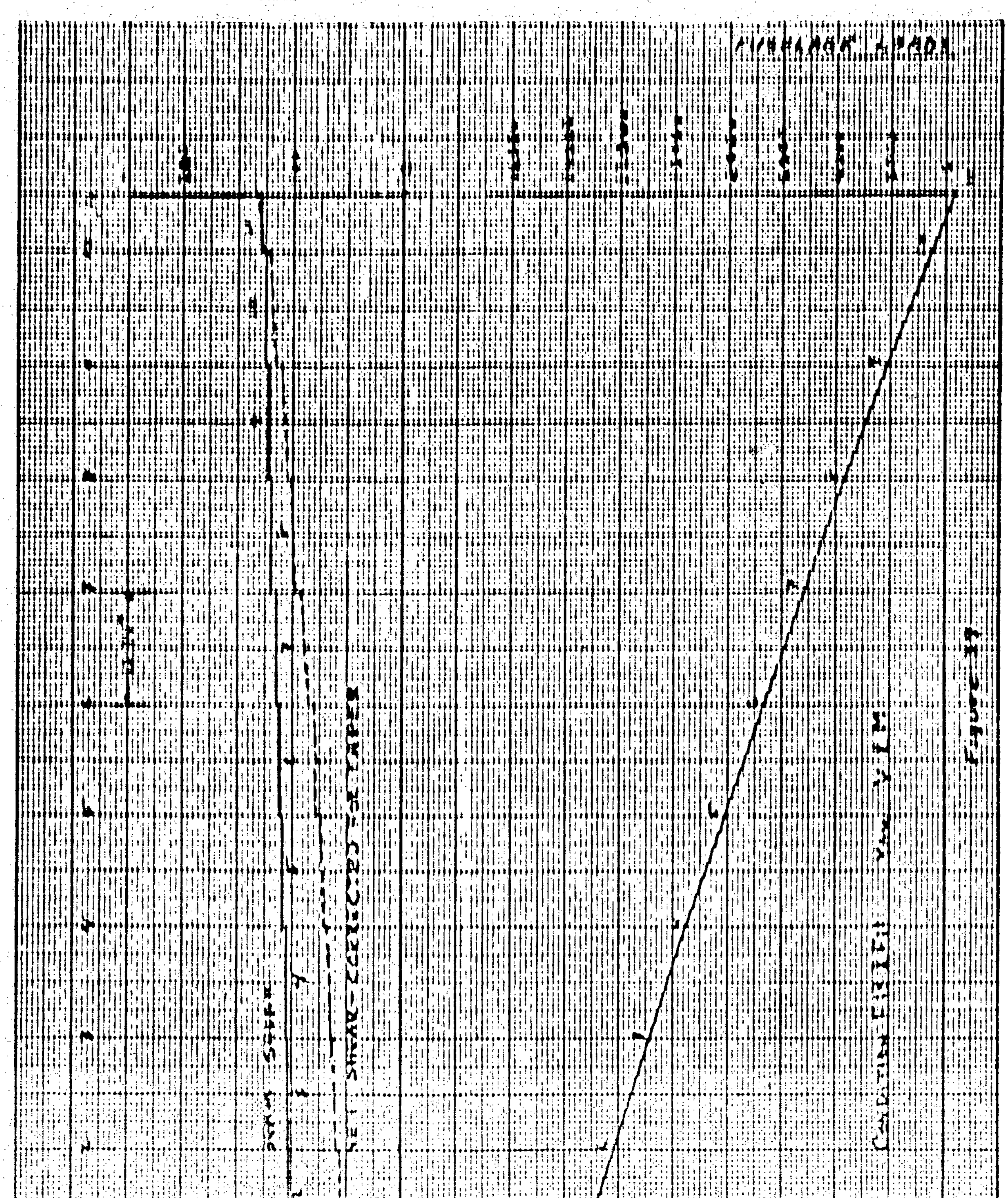

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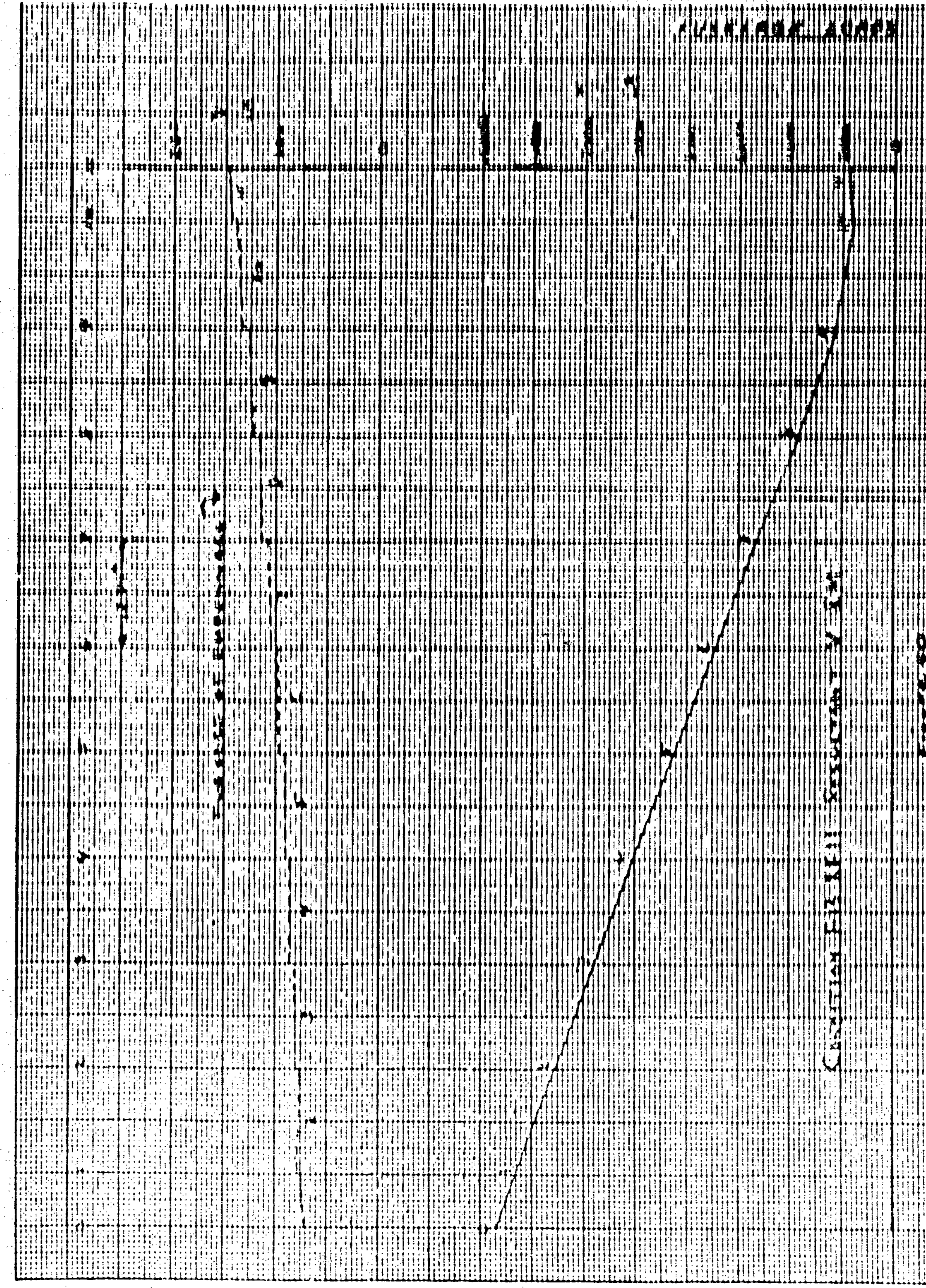


ENGINE MOUNT LOADS

REF. DWG 447A-007

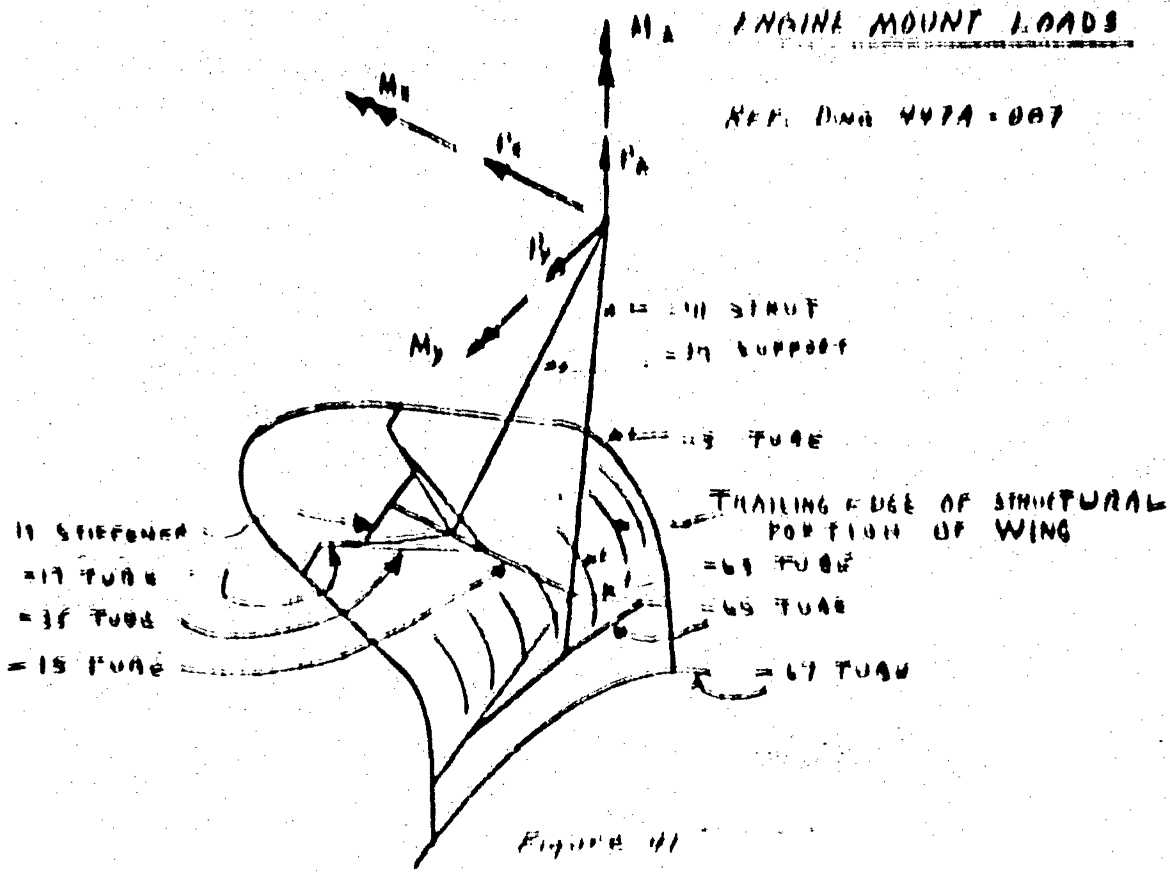


Figure 41

CRITICAL LOADS RESOLVED AT TOP OF TRUSS

Table XX

CONDITION	REF.	P_x LB	P_y LB	P_z LB	M_x IN-LB	M_y IN-LB	M_z IN-LB
$n_z = -1$ FLIGHT	PG. 2.04.010	915	0	-187	1571	-2625	0
$n_z = -3$ FLIGHT	PG. 2.04.010	230	0	-273	1572	-1856	0
$n_y = 1.31$ SIDE LOAD	REF. 7	230	-99	-79.5	1180	-1170	-456

CRITICAL ENGINE MOUNT LOADS

EMPENNAGE LOADS

The critical loading condition alone is presented as the stress analysis is a summary one. This load is 101 lb acting on the vertical tail. The distribution, using a 100 lb unit load, is that shown on page 2.03.070.

The hinge line reactions and geometry for the rudder and vertical stabilizer are given on pages 2.05.020 and 2.05.030. As the rudder attachment is statically determinate, the calculations are given on page 2.05.020. The support of the vertical stabilizer is statically indeterminate as is apparent from the equations below. (See page 2.05.030).

$$F_x = F_x - A_x - 32.2 + R_x = 0$$
$$F_y = -F_y - A_y - 7.45 + R_y = 0$$
$$F_z = F_z + A_z - 100 + R_z = 0$$

$$M_x = 24.1\ F_z + 30.6\ A_z - 35.2 \times 19.5 - 64.0 \times 10.22 = 0$$
$$M_y = 29\ F_z + R_z\ X_{Rz} - 64.0 \times 11.16 = 0$$
$$M_z = -24.1F_x + 29F_y + 30.6A_x + 32.2 \times 22 - R_y\ X_{Ry} = 0$$

With direction cosines of page 2.05.030.

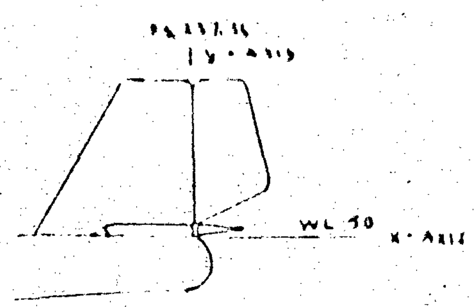
$$.264F - .102A + R_x = 35.2$$
$$-.690F - .612A + R_y = 7.45$$
$$.675F + .702A + R_z = 100$$
$$16.25F + 23.7A = 1870$$
$$19.57F + X_{Rz} = 723$$
$$13.64F + 3.12A - X_{Ry} = -708.4$$

These are plotted on Pg. 2.05.040

Solution when $X_{Rz} = X_{Ry}$

$F = .038A = 2.92$ lb
$A = 76.7$ lb
$R_x = 39.3$ lb
$R_y = 55.9$ lb
$R_z = 30.5$ lb
$X_{Ry} = X_{Rz} = 17.6$ In.
$F_z = 1.97$ lb $A_y = 47.0$ lb
$F_y = 2.01$ lb $A_z = 60.0$ lb

The shear and moment diagrams are on page 2.05.050.



SKETCH SHOWING ORIENTATION OF AXES

EMPENNAGE LOADS

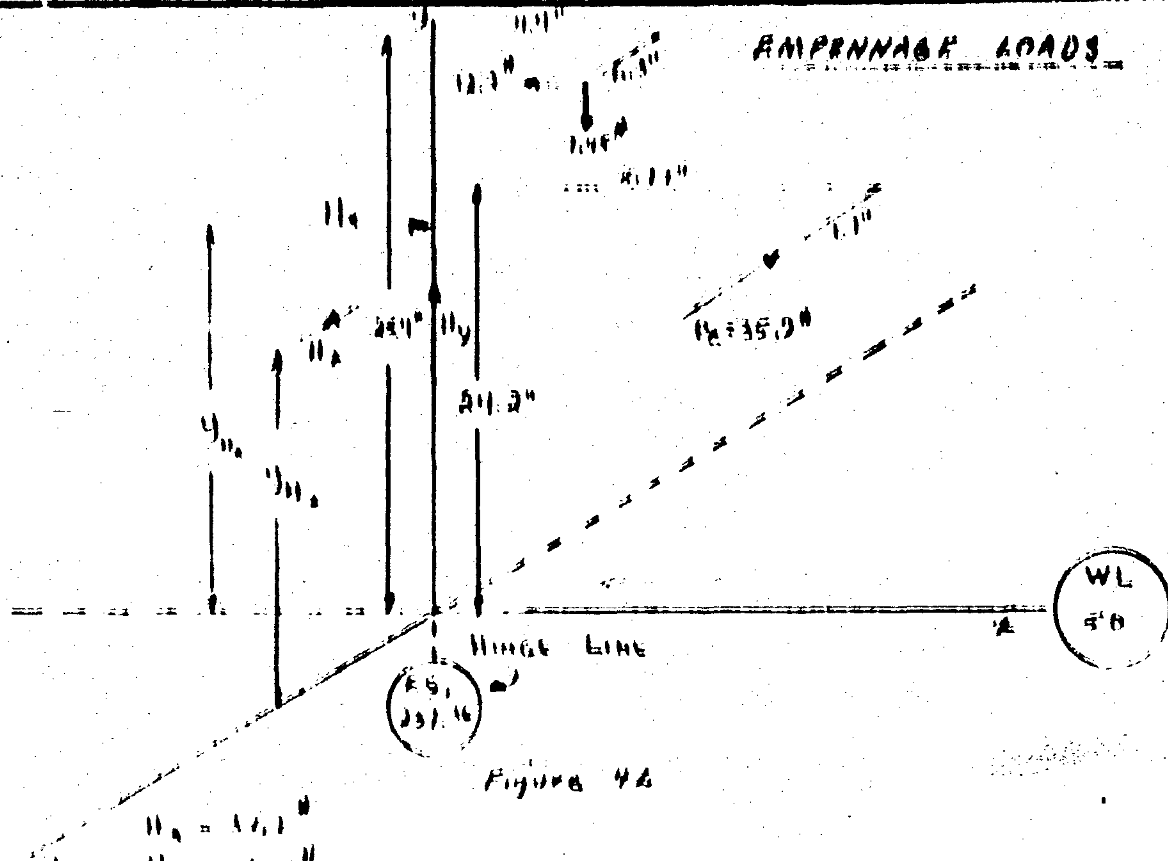


Figure 4A

H_x = [illegible]"
H_y = [illegible]"
H_z = [illegible]"

ΣM_x = [illegible] y_{H_z} − [illegible]([illegible]) + [illegible]([illegible]) = 0 y_{H_z} = [illegible]"

ΣM_y = [illegible] = 0 [illegible]

ΣM_z = − [illegible] y_{H_x} + [illegible]([illegible]) − [illegible]([illegible]) = 0 y_{H_x} = 24.9"

HINGE LINE REACTION FOR 100 LB ON VERTICAL TAIL

EMPENNAGE LOADS

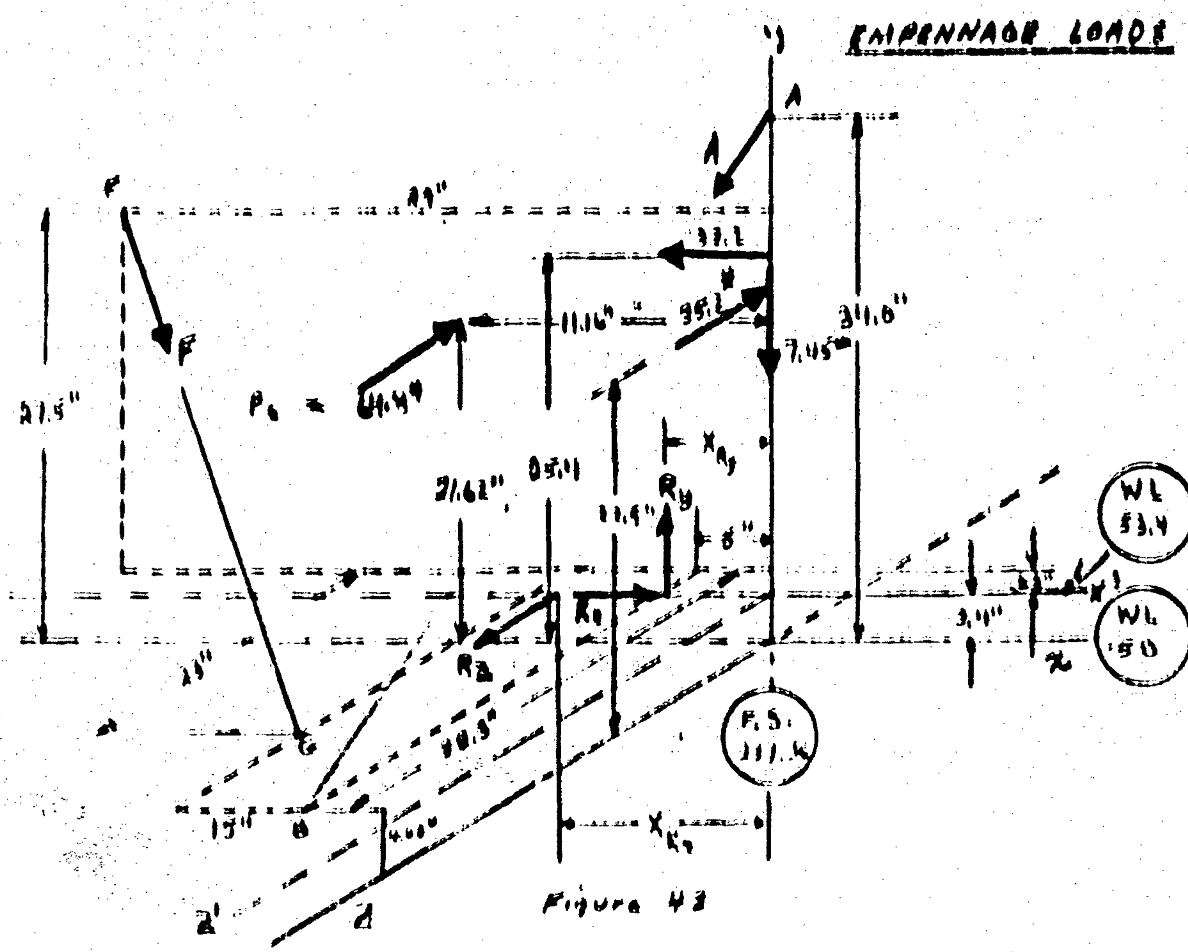


Figure 43

LOADS AND REACTIONS ON VERTICAL STABILIZER FOR 100 LB. SIDE LOAD.

DIRECTION NUMBERS AND COSINES FOR CABLE TENSIONS F AND A.

$x_F = 29 - 5 - 15 = 9"$	$x_A = 5"$
$y_F = 27 - 3.4 - .68 = 23.5"$	$y_A = 34 - 4.03 = 30"$
$z_F = 23"$	$z_A = 38.3"$
$FG = \sqrt{9^2 + 23.5^2 + 23^2} = 34.1"$	$AB = \sqrt{5^2 + 30^2 + 38.3^2} = 49.0"$
$F_x = \frac{9}{34.1} F = .274 F$	$A_x = \frac{5}{49} A = .102 A$
$F_y = \frac{23.5}{34.1} F = .690 F$	$A_y = \frac{30}{49} A = .612 A$
$F_z = \frac{23}{34.1} F = .675 F$	$A_z = \frac{38.3}{49} A = .781 A$

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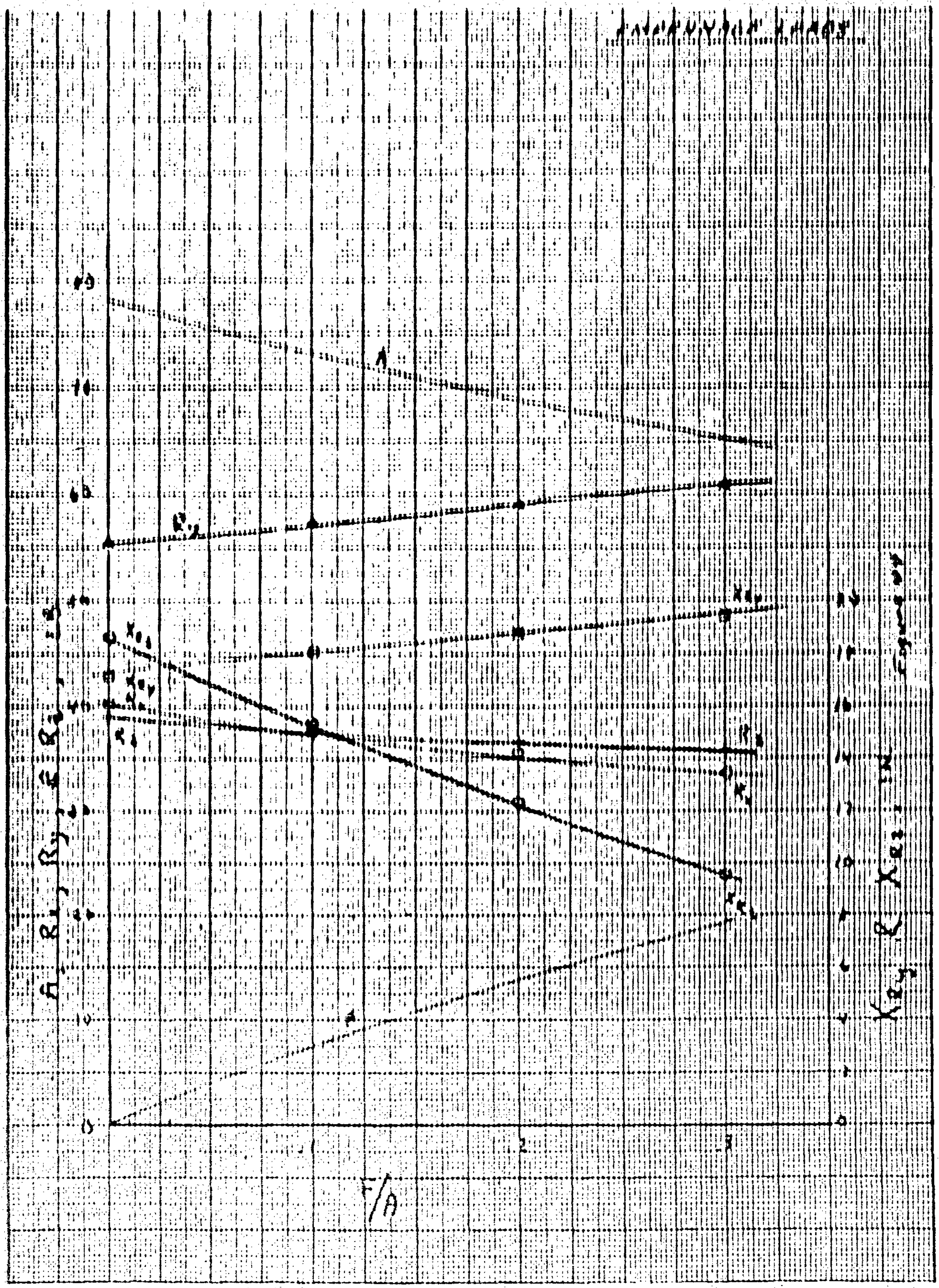

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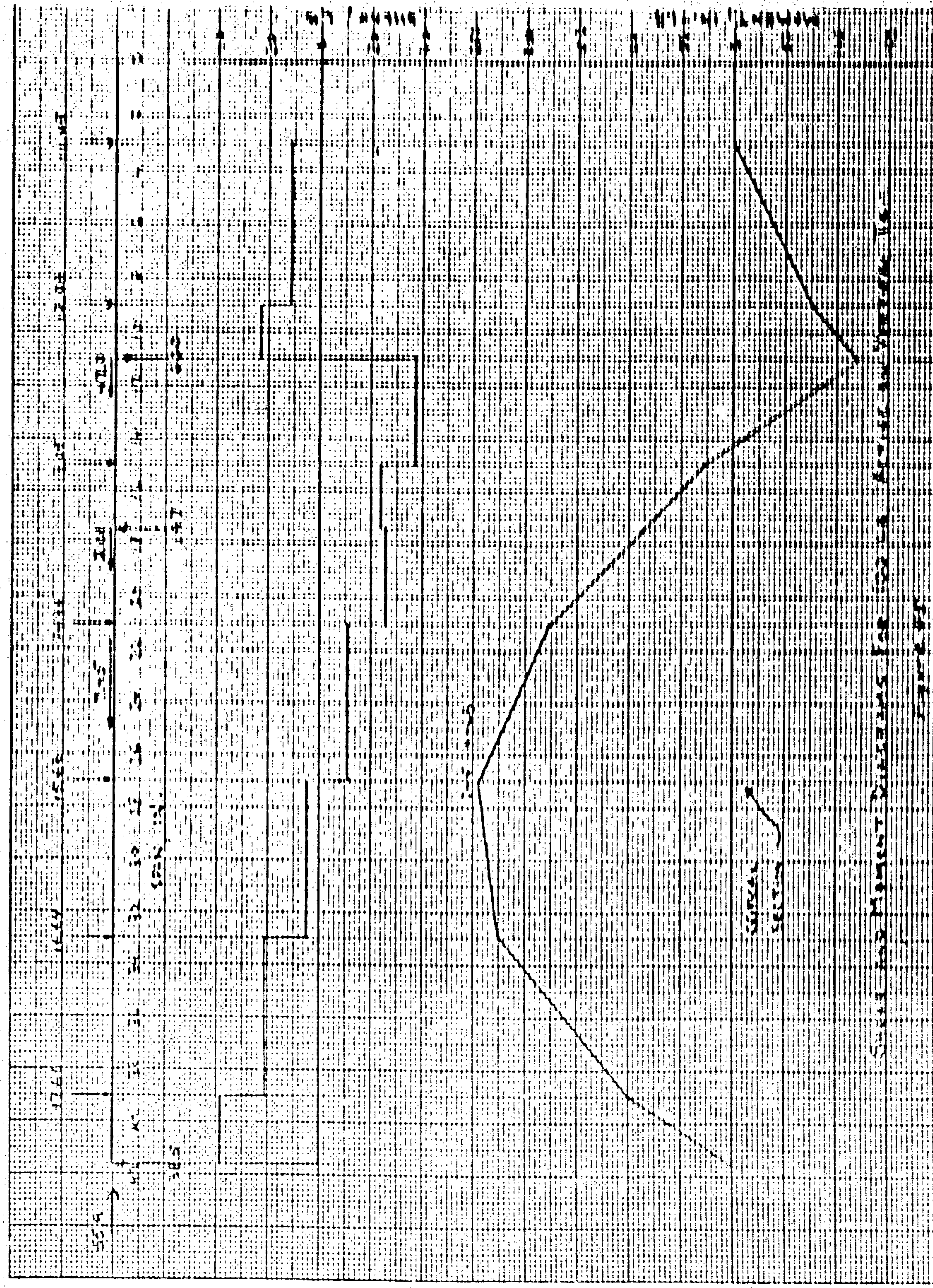

EMPENNAGE LOADS

TORQUE AT THE CRITICAL SECTION IS CALCULATED BELOW.

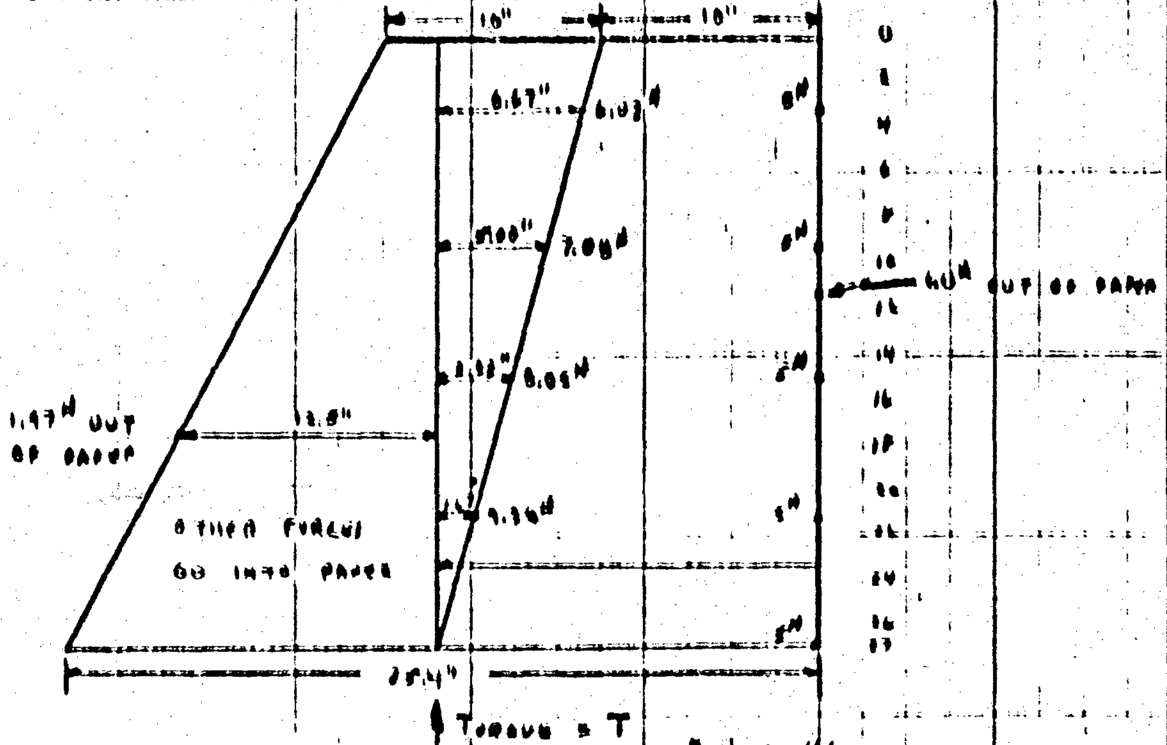

Figure 46

$T = 40 \times 17.7 - 1.97 \times 12.5 - 6.67 \times 6.03 - 5.00 \times 7.04 - 3.33 \times 8.04 - 1.67 \times 9.38 - 17.7 \times 5$
$= 477$ IN-LB

CORRECTION FOR INERTIA OF VERTICAL TAIL

THE VERTICAL TAIL WEIGHT 5.4 LB AND ITS CENTROID IS AT WL 73.3, STA 330.6 (PG 2.01.070). THE INERTIA LOAD DUE YAWING ACCELERATION IS:

$$P = \frac{W}{g} \times r\ddot{\psi} = \frac{5.4}{32.2}(13.13)(8) = 17.6 \text{ LB}$$

WHERE $r = (330.6 - 172.56) \div 12 = 13.13'$ } SEE PG. 2.03.220
$\ddot{\psi} = 8$ RAD/SEC^2

THUS THE ACTUAL TAIL LOAD OF 181# MAY BE REDUCED BY 17.6 LB TO 163 LB.

Cockpit Loads

The cockpit loads are:

(1) Pullout $n_z = 2.5$

(2) Pushover $n_z = 1.0$

(3) Yaw $n_z = 1$, $n_y = .675$

(4) Roll $n_z = 1$, $M_{roll} = 730$ in-lb, $M_{yaw} = 1537$ in-lb.

(5) Level Landing, inclined reactions

$$n_z = \frac{1347}{550} - \frac{1}{3} + 1 = 3.12$$

$$n_x = -\frac{371}{550} = -.675$$

(6) Tail Down Landing $n_z = 3.12$

(7) Ground Load $n_z = 1.33$, $n_y = .83$

Conditions (1) - (4) are flight conditions and are partially derived from Table III, page 2.00.030.

Conditions (5) - (7) are derived from cam -03, Art. 3.243 to 3.249.

The most critical loads are (1), (5), and (6).

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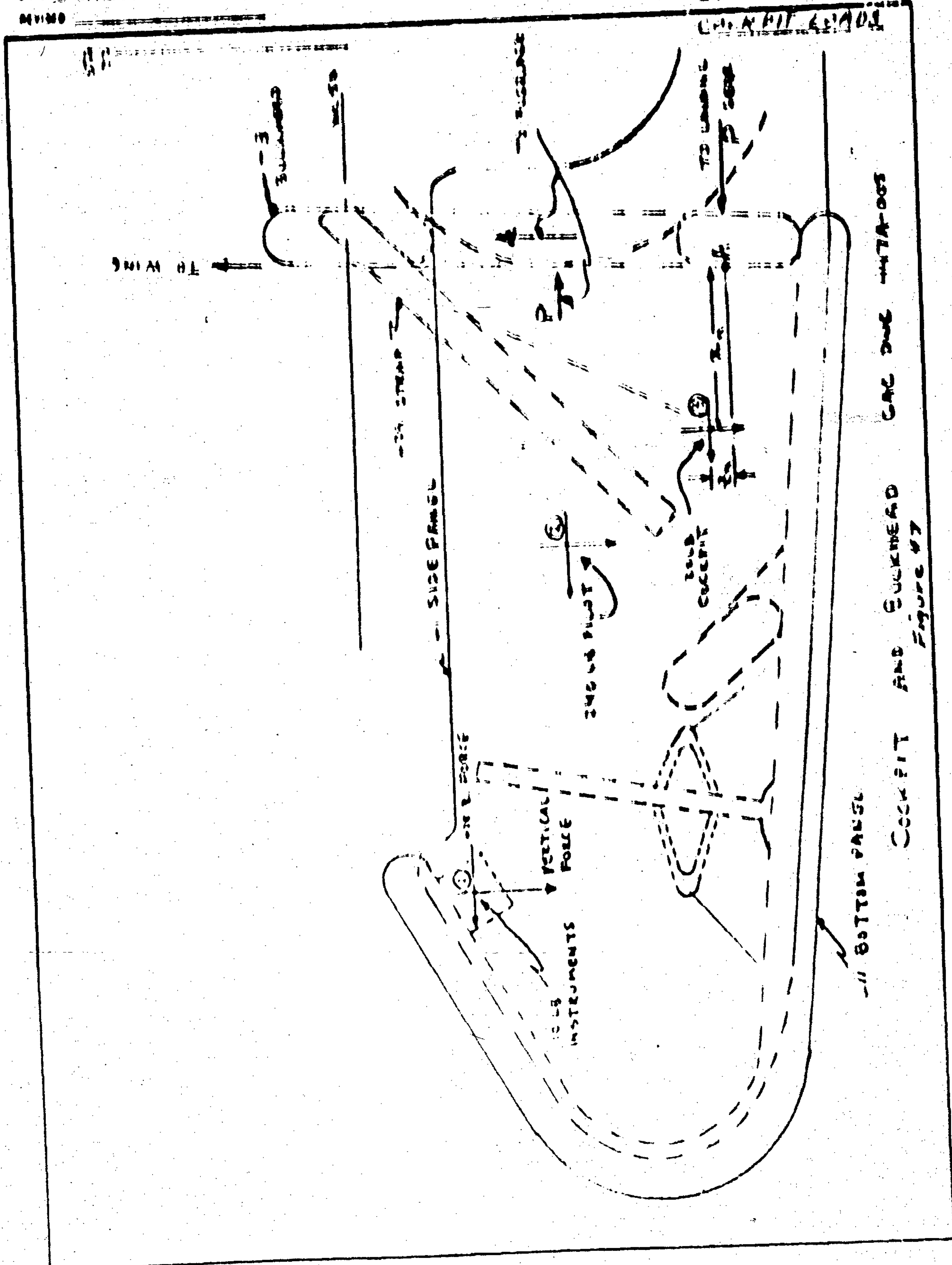

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COCKPIT LOADS

Position of loads (1), (2), and (3) see cockpit and bulkhead sketch. The positions are measured from a point 'A' on the line of action of the reaction between the landing gear and the bottom and side panels of the cockpit. Actually point 'A' is dependent upon the load factor, however an uppermost position was determined by calculations using static test results.

Position of Loads

Coordinate	Instruments (1)	Pilot (2)	Cockpit (3)
X	37"	16.2"	9.8"
Z	17"	10.6"	1.6"

Table XXI
Inertia Loads

Condition	Load (1); 10# Instru.		Load (2); 200# Pilot		Load (3); 20# Cockpit	
	Horizontal 10 N_x, LB	Vertical 10 N_z, LB	Horizontal 200 N_x, LB	Vertical 200 N_z, LB	Horizontal 20 N_x, LB	Vertical 20 N_z, LB
1	0	25	0	600	0	50
5	6.75	31.2	16.2	740	13.5	62.4
6	0	31.2	0	740	0	62.4

Table XXII
Moments at Limit Loads

Condition	(1)		(2)		(3)		Sum
	Horiz IN-LB	Vert. IN-LB	Horiz IN-LB	Vert. IN-LB	Horiz. IN-LB	Vert IN-LB	IN-LB
1	0	925	0	9720	0	490	11,125
5	115	1155	1720	12160	22	611	15,773
6	0	1155	0	12150	0	611	13,916

Moment of horiz. force = horiz force times Z coordinate,
" " vert " = vert " " X " .

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Landing Gear Loads

The CAM 3, reference 7, was used as a guide in the determination of landing loads for this aircraft. Article 3.243 of reference 7 gives a minimum descending velocity of 7 ft/sec. However, preliminary investigation showed that such a minimum could not be achieved without an elaborate and heavy landing gear. It was found that the kinetic energy of a 5 ft/sec descending velocity could be absorbed by a single wheel attached to a piston deflecting into the fuselage. The load and energy vs. deflection characteristics of the piston-fuselage system are given in figures 48 and 49, while figures 50 and 51 are for the tire. See reference 8 for the piston-fuselage deflection characteristics.

The conditions examined with the limit loads are given below. One flight condition is included as the lower wing brace cables are attached to the landing gear.

TABLE XXIII

Condition	Limit Loads, Lb.		
	Vertical	Aft	Side
(1) Level Landing, Inclined Reactions	1347	371	0
(2) Tail-Down Landing	1347	0	0
(3) Side Load	732	0	457
(4) Flight Load, $n_z = 3$	1353/1.5 = 902		

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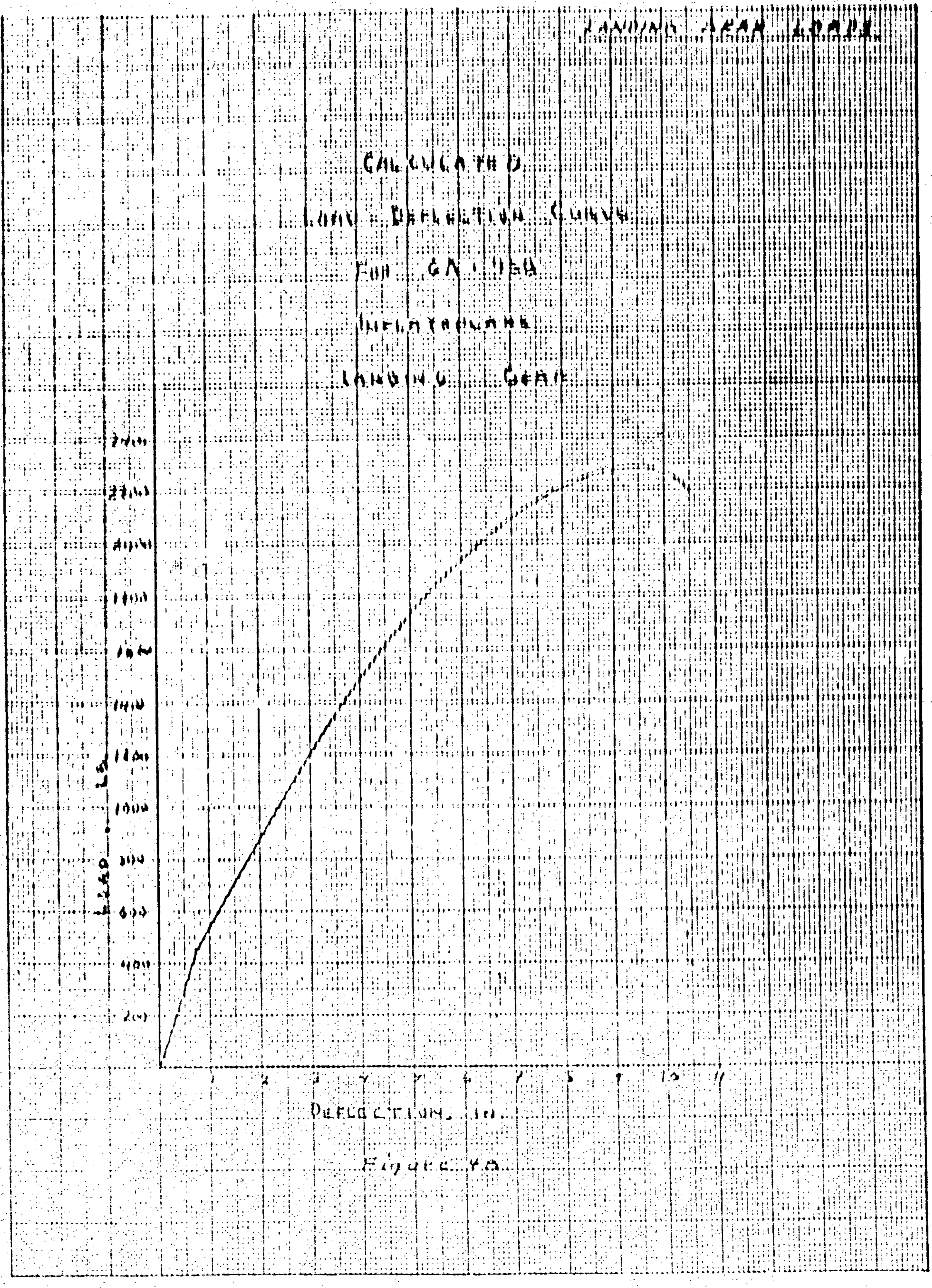
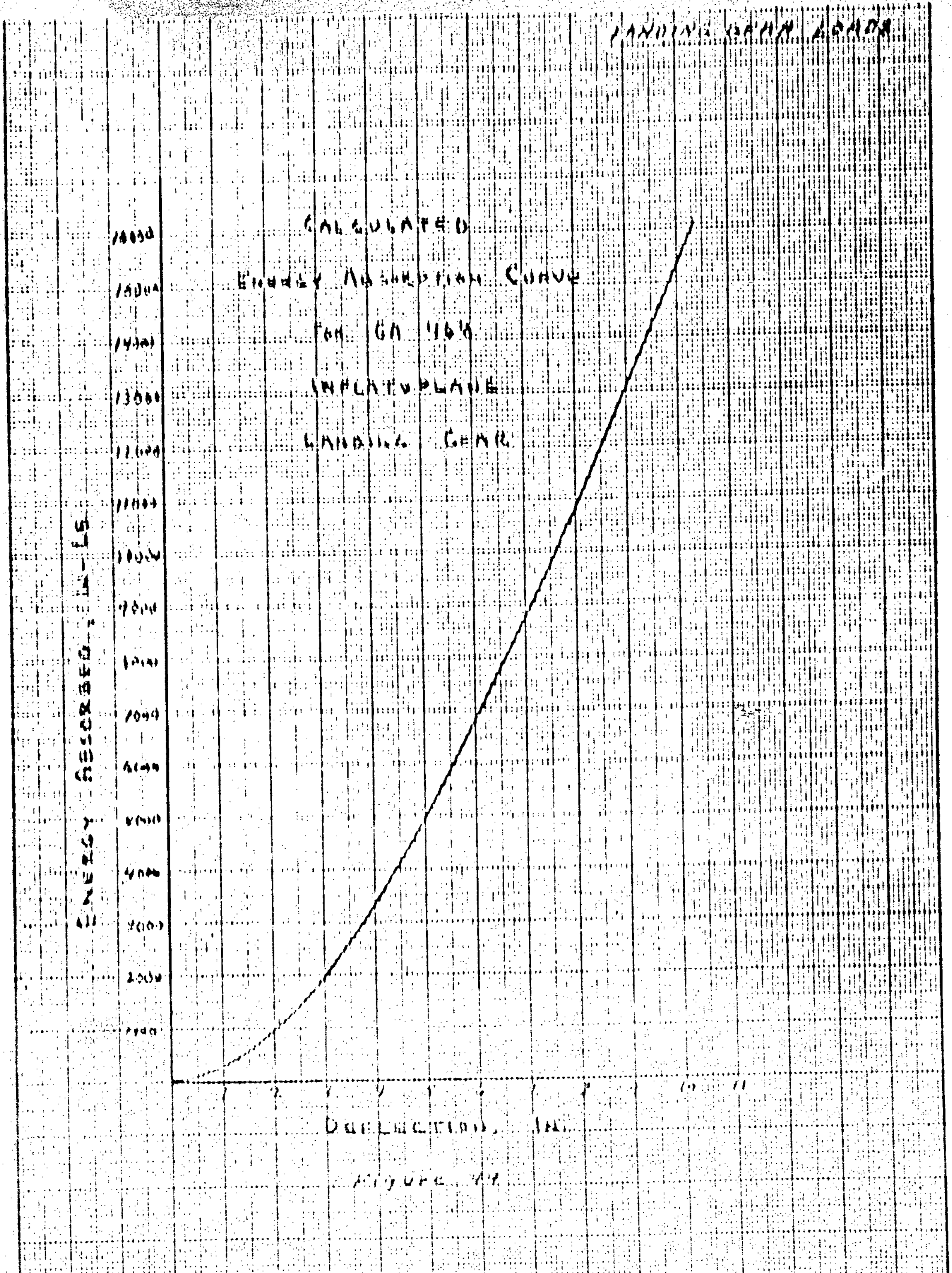


Figure 48


LANDING GEAR LOADS
CALCULATED
ENERGY ABSORPTION CURVE
FOR GA 468
INFLATOPLANE
LANDING GEAR
ENERGY ABSORBED - IN-LBS
DEFLECTION, IN.

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LANDING GEAR LOADS

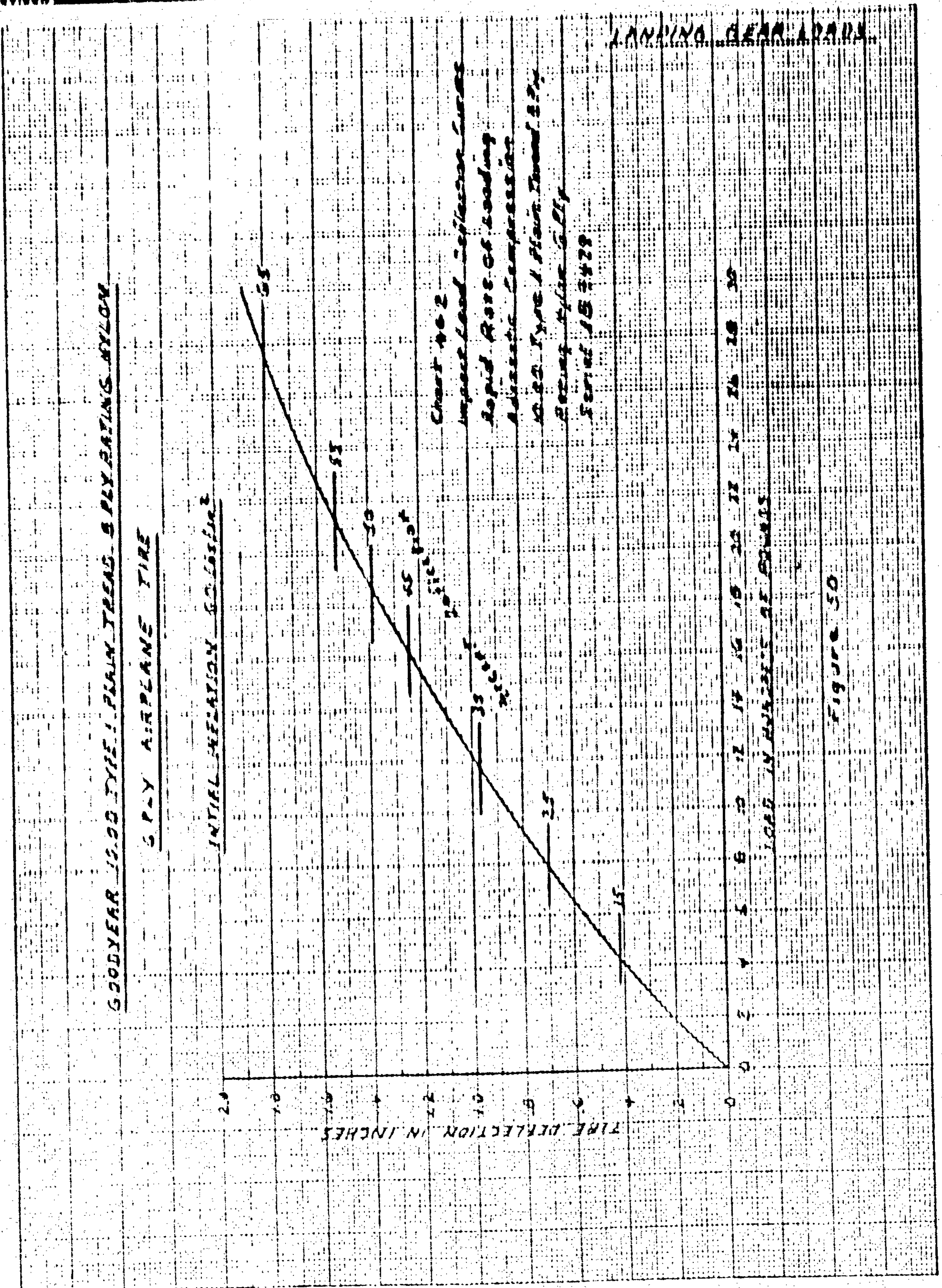

Figure 50

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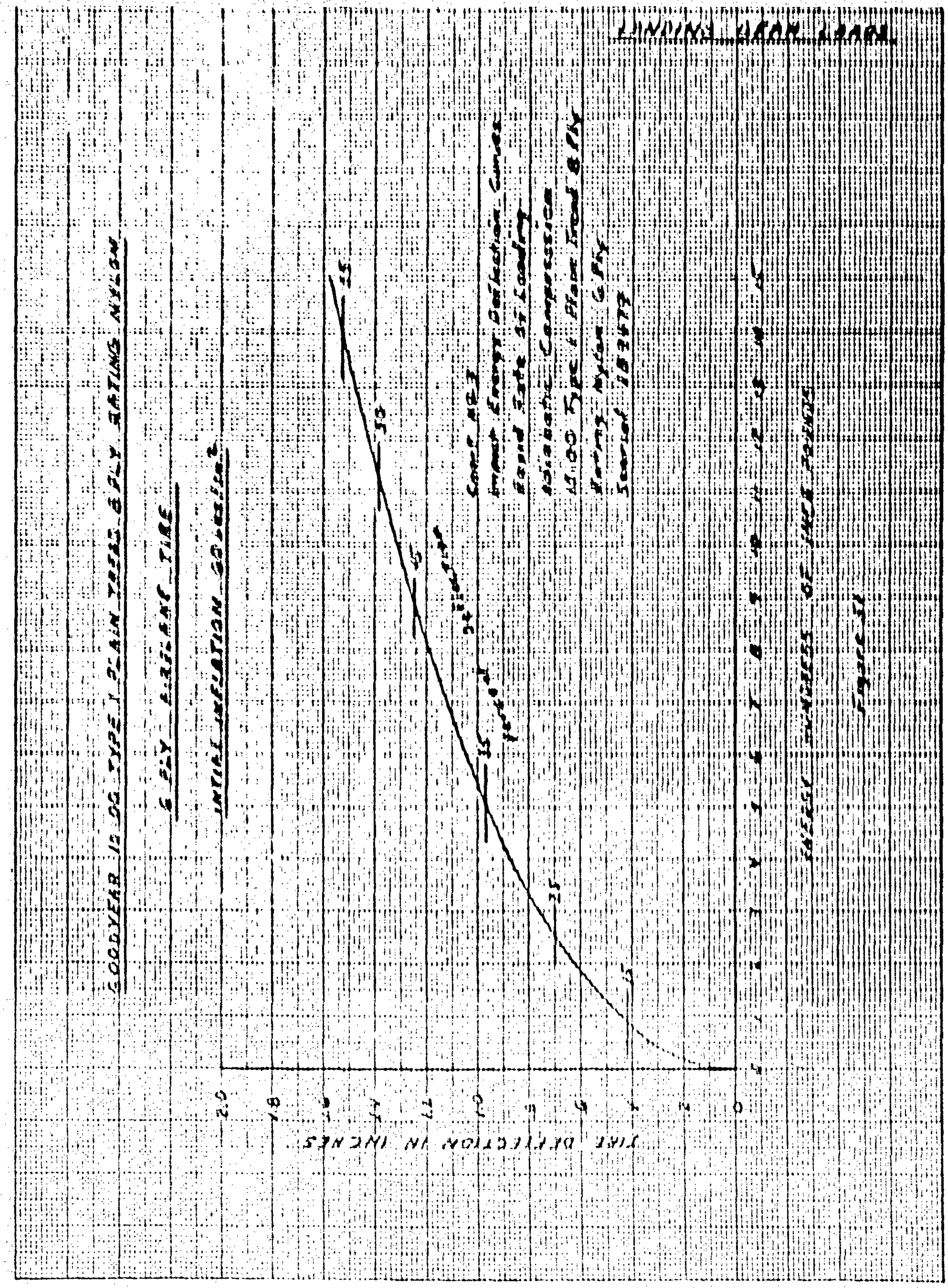

WING ANALYSIS

Section 3

WING ANALYSIS

The wing is of Airmat construction and supports the external loads by virtue of tensile inflation stress. If one of the principal stresses at a point on the wing cross-section becomes zero due to the applied compressive stresses a wrinkle starts to form there. As more load is applied the wrinkle will enlarge to a point where collapse will occur. The pressure in the wing should be large enough to prevent a wrinkle from forming under limit loads and collapse at ultimate loads. For this analysis the ultimate load is 1.75 times limit load.

While the minimum principal stress at the point where a wrinkle first forms is zero, both principal stresses at a point on the opposite surface of the wing are tensile stresses. The allowable strength value on the tension side is the quick break value derived from a cylinder burst test divided by a creep rupture factor of 3. The factor of 3 accounts for the fact that fabric under load for a period of time has a reduction in strength.

The allowable hoop tension strength value on the compression side is the quick break value derived from the cylinder burst test times 0.65 divided by 3. The factor of 0.65 accounts for the fact that the fabric in a cylinder burst test is loaded at a 2-1 stress ratio, while on the compression side of the wing is loaded in only one direction and consequently gets little or no help from the bias plies. The factor is based on a comparison of cylinder burst tests and strip tensile tests.

Since the negative margins shown in Tables XXXII and XXXIII, pages 3.01.220 and 3.01.230 respectively, cover only a small percentage of the periphery of the wing cross-section shown on page 3.01.040, these negative margins are not serious. This is further proved in the wind tunnel test, which gave an ultimate margin of safety of +0.17, reference page 3.01.240.

WING ANALYSIS

Calculation of Bending Stress

Method of Analysis

For a cross-section in bending, the general equation for stresses due to bending may be determined from the following expression:

$$f_b = (C_1 M_z - C_2 M_x)\, z + (C_1 M_x - C_3 M_z)\, x$$

Where

$$C_1 = \frac{I_{xz}}{I_x I_z - I_{xz}^2}$$

$$C_2 = \frac{I_z}{I_x I_z - I_{xz}^2}$$

$$C_3 = \frac{I_x}{I_x I_z - I_{xz}^2}$$

I_x = Moment of inertia of cross-section about x-x axis
I_z = Moment of inertia of cross-section about z-z axis
M_x = Bending moment about x-x axis, positive when it causes compression above x-x axis
M_z = Bending moment about z-z axis, positive when it causes compression to right of z-z axis

x and z are the ordinates of the elements of the cross-section. x is positive above the x-x axis and z is positive to the right of the z-z axis.

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Section Properties

Table XXIV

Col.	①	②	③	④	⑤	⑥	⑦	⑧
Element	Area	Z'	AZ'	AZ'^2	X'	AX'	AX'^2	AX'Z'
Ref.			① × ②	② × ③		① × ⑤	⑤ × ⑥	⑥ × ⑤
1	0.85	0.53	0.4505	0.2388	0.11	0.0935	0.0103	0.0496
2	1.55	1.48	2.294	3.3951	0.82	1.271	1.342	1.9811
3	1.35	2.20	2.970	6.534	1.96	2.646	5.186	5.8212
4	2.40	2.90	6.960	20.184	3.68	8.832	32.502	25.6128
5	4.70	3.71	17.437	64.691	7.13	33.511	238.933	124.326
6	4.56	4.28	19.517	83.533	11.72	53.443	626.352	228.739
7	4.61	4.50	20.745	93.353	16.31	75.191	1226.33	338.351
8	4.61	4.47	20.607	92.113	20.90	96.349	2013.69	430.686
9	4.59	4.25	19.508	82.909	25.50	117.045	2984.65	497.454
10	4.58	3.96	18.137	71.823	30.09	137.812	4146.76	545.742
11	4.62	3.58	16.540	59.213	34.68	160.222	5556.50	573.631
12	4.61	3.07	14.153	43.450	39.27	181.035	7109.24	555.773
13	4.65	2.49	11.579	28.832	43.86	203.949	8945.20	507.835
14	1.04	2.04	2.122	4.329	46.59	48.453	2241.96	98.804
15	0.62	1.57	0.9734	1.528	47.26	29.301	1384.77	46.003
16	1.40	0.90	1.260	1.134	47.71	66.794	3186.74	60.115
Σ Upper	50.74		175.25	657.26		1215.95	40111.37	4040.7
Σ Lower	50.74		-175.25	657.26		1215.95	40111.81	-4040.7
Total	101.48		0	1314.52		2431.90	80223.14	0

$\bar{X} = \frac{\Sigma AX'}{\Sigma A} = \frac{2431.9}{101.48} = 23.95$ in.

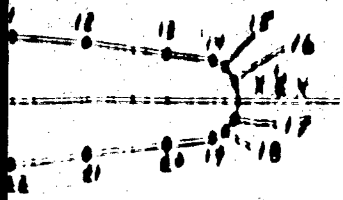
$\bar{Z} = \frac{\Sigma AZ'}{\Sigma A} = \frac{0}{101.48} = 0$

$I_{Z'} = \Sigma AX'^2 + I_{0_Z} = 80$

$I_Z = I_{Z'} - \bar{X}^2 \Sigma A = 80{,}550 - (23.95^2 \times 101.48) = 22{,}[illegible]$ in.^{3}

$I_X = \Sigma AZ'^2 - \bar{Z}\Sigma A = 1314.52 - 0 \times 101.48 = 1315$ in.^{3}

WING ANALYSIS



XXIV

(6)	(7)	(8)	(9)	(10)	(11)
Ax'	Ax'^2	$Ax'z'$	I_{o_z}	Distance From N.A. To Outer Fiber	
				x	z
(1)x(5)	(5)x(6)	(2)x(5)	$\frac{1}{12}bh^3$	(5) - $\bar{x}$	(2) - $\bar{z}$
0.0735	0.0103	0.0410	—	-23.84	3.53
1.271	1.342	1.3911	—	-23.13	1.48
2.646	5.186	5.8212	0.205	-21.77	2.23
3.812	31.503	25.6128	1.15	-20.27	2.70
33.511	213.913	124.126	8.15	-16.82	3.71
51.441	620.652	220.719	7.72	-12.21	4.23
75.137	1226.33	333.351	9.16	-7.64	4.50
96.841	2313.69	430.682	8.16	-3.05	4.47
117.045	2784.65	477.454	8.07	1.55	4.25
137.812	4146.75	545.742	8.04	6.14	3.96
163.222	5556.50	573.637	8.21	10.75	3.53
181.035	7107.24	555.713	8.16	15.32	3.07
203.744	8745.20	507.855	8.41	19.91	2.47
49.453	2741.96	98.364	—	22.64	2.04
27.301	1394.77	46.008	—	23.31	1.57
66.714	3186.14	60.115	—	28.16	0.93
1215.95	40199.57	4040.9	75.135		
1215.95	40199.57	-4040.9	75.135		
2431.90	80399.14	0	150.270		

$I_x' = \Sigma Ax'^2 + I_{o_z} = 80399.14 + 150.27 = 80,550 \text{ in}^4$

$I_z = 22,341 \text{ in}^4$

$I_{xz} = 0$

$K = I_x I_z - I_{xz}^2 = 1315(22,341) - 0$
$= 29.3784(10^6) \text{ in}^8$

$C_1 = I_{xz}/K = \frac{0}{29.3784(10^6)} = 0$

$C_2 = I_z/K = \frac{22,341}{29.3784(10^6)} = 760.453(10^{-6}) \text{ in}^{-4}$

$C_3 = I_x/K = \frac{1315}{29.3784(10^6)} = 44.76(10^{-6}) \text{ in}^{-4}$

Note:

For Lower Surface Change Sign in z Column (11) For Elements 17 Thru 22

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WING ANALYSIS

Calculation of Bending Stress

Condition A_2 Symmetrical Maneuver

Wing Sta. 0.00 (Wing Root)

$$f_b = (C_1 M_z - C_2 M_x) z + (C_1 M_x - C_3 M_z) x$$

$C_1 = 0$ $M_x = 3400$ in.-lbs. Ref. Pg. 2.02.180

$C_2 = 760.456 \times 10^{-6}$ $M_z = -1310$ in.-lbs. Ref. Pg. 2.02.180

$C_3 = 44.761 \times 10^{-6}$

$$(C_1 M_z - C_2 M_x) = -(760.456 \times 10^{-6})(3400)$$

$$= -2.59$$

$$(C_1 M_x - C_3 M_z) = (44.761 \times 10^{-6})(1310)$$

$$= 0.0586$$

$\therefore$

$$f_b = 0.0586\, x - 2.59\, z$$

Prepared by J. [illegible]
Checked by [illegible]
Date 1-19-81
Revised

GOODYEAR AIRCRAFT

Page 1.01.060
Model [illegible]
Ser. [illegible]
Rep. No. [illegible]

Bending f_b, Wing Sta 0.00, [illegible] XXV WING ANALYSIS

Gal Element	(ii) X	(iii) Z	(iv) = 2.59 Z	(v) 0.0594 X	(vi) f_b = (iii) + (iv)	(vii) (i) × (vi)
1	-23.84	0.53	-1.38	-1.40	-2.78	-2.34
2	-23.13	1.48	-3.83	-1.36	-5.19	-8.05
3	-21.79	2.20	-5.70	-1.29	-6.99	-9.44
4	-20.27	2.90	-7.52	-1.19	-8.71	-20.90
5	-16.82	3.11	-9.60	-0.99	-10.59	-49.73
6	-12.23	4.28	-11.10	-0.72	-11.82	-54.03
7	-7.64	4.50	-11.68	-0.45	-12.13	-56.63
8	-3.05	4.41	-11.60	-0.18	-11.78	-54.28
9	1.55	4.25	-11.00	0.09	-10.91	-50.03
10	6.14	3.76	-10.25	0.36	-9.89	-45.33
11	10.73	3.58	-9.29	0.63	-8.66	-40.03
12	15.32	3.01	-7.95	0.90	-7.05	-32.58
13	19.91	2.49	-6.45	1.17	-5.28	-24.40
14	22.64	2.04	-5.30	1.33	-3.97	-8.13
15	23.31	1.57	-4.07	1.37	-2.7	-1.68
16	[illegible].16	0.90	-2.33	1.39	-0.94	-1.32
17	23.16	0.90	2.33	1.39	3.72	5.22
18	23.31	-1.57	4.07	1.37	5.44	3.37
19	22.64	-2.04	5.30	1.33	6.63	8.90
20	19.91	-2.49	6.45	1.17	7.62	35.41
21	15.32	-3.01	7.95	0.90	8.85	40.81
22	10.73	-3.58	9.29	0.63	9.92	45.86
23	6.14	3.76	10.25	0.36	10.61	48.57
24	1.55	-4.25	11.00	0.09	11.09	50.87
25	-3.05	-4.41	11.60	-0.18	11.42	52.64
26	-7.64	-4.50	11.68	-0.45	11.23	51.82
27	-12.23	-4.28	11.10	-0.72	10.38	47.31
28	-16.82	-3.11	9.60	-0.99	8.61	40.50
29	-20.27	-2.90	7.52	-1.19	6.33	15.20
30	-21.79	-2.20	5.70	-1.29	4.41	5.95
31	-23.13	-1.48	3.83	-1.36	2.47	3.83
32	-23.84	-0.53	1.38	-1.40	-0.02	-0.02
Σ					Stress check →	0.00

219-83 (3-57)

WING ANALYSIS

Calculation of Bending Stress

Condition C_2 Symmetrical Maneuver

Wing Sta. 0.00 (Wing Root)

$$f_b = (C_1 M_z - C_2 M_x)\, z + (C_1 M_x - C_3 M_z)\, x$$

$C_1 = 0$ $\qquad$ $M_x = 3945$ in.-lbs.

$C_2 = 760.456 \times 10^{-6}$ $\qquad$ $M_z = 2010$ in.-lbs.

$C_3 = 44.761 \times 10^{-6}$ $\qquad$ Loads Ref. Pg. 2.01.030 & 2.01.050

$$(C_1 M_z - C_2 M_x) = -(760.456 \times 10^{-6})(3945)$$
$$= -3$$

$$(C_1 M_x - C_3 M_z) = -(44.761 \times 10^{-6})(2010)$$
$$= -0.13$$

$\therefore$

$$f_b = -0.13\, x - 3\, z$$

Prepared by JCH
Checked by
Date 1-14-41
Revised

GOODYEAR AIRCRAFT

Page 5.01.080
Model GA-440
Ser. 9861
Rep. No. S 07-3

Condition G.R.-I Wing Sta. 0.00 Table XXVI WING ANALYSIS

Col.	(11)	(12)	(12)	(13)	(14)	(15)
Element	X	Z	-3 Z	-0.13 X	f = (12) + (13)	(7) × (14)
1	-23.84	0.53	-1.59	3.10	1.51	1.29
2	-23.13	1.48	-4.44	3.01	-1.43	-2.22
3	-21.99	2.20	-6.60	2.86	-3.74	-5.05
4	-20.27	2.90	-8.70	2.64	-6.06	-14.55
5	-16.82	3.71	-11.13	2.19	-8.94	-42.00
6	-12.23	4.28	-12.84	1.59	-11.25	-51.40
7	-7.64	4.50	-13.50	1.00	-12.50	-57.60
8	-3.05	4.47	-13.41	0.40	-13.01	-60.00
9	1.55	4.25	-12.75	-0.20	-12.95	-59.50
10	6.14	3.96	-11.88	-0.80	-12.68	-58.00
11	10.73	3.58	-10.74	-1.40	-12.14	-55.90
12	15.32	3.07	-9.21	-1.99	-11.20	-51.70
13	19.91	2.49	-7.47	-2.59	-10.06	-46.80
14	22.64	2.04	-6.12	-2.95	-9.07	-9.45
15	23.31	1.57	-4.71	-3.03	-7.74	-4.80
16	23.76	0.90	-2.70	-3.09	-5.79	-8.10
17	23.76	-0.90	2.70	-3.09	-0.39	-0.55
18	23.31	-1.57	4.71	-3.03	1.68	1.00
19	22.64	-2.04	6.12	-2.95	3.17	3.30
20	19.91	-2.49	7.47	-2.59	4.88	22.30
21	15.32	-3.07	9.21	-1.99	7.22	33.30
22	10.73	-3.58	10.74	-1.40	9.34	43.10
23	6.14	-3.96	11.88	-0.80	11.08	50.70
24	1.55	-4.25	12.75	-0.20	12.55	57.60
25	-3.05	-4.47	13.41	0.40	13.81	63.90
26	-7.64	-4.50	13.50	1.00	14.50	67.00
27	-12.23	-4.28	12.84	1.59	14.43	66.00
28	-16.82	-3.71	11.13	2.19	13.32	62.60
29	-20.27	-2.90	8.70	2.64	11.34	27.19
30	-21.99	-2.20	6.60	2.86	9.46	12.80
31	-23.13	-1.48	4.44	3.01	7.45	11.55
32	-23.84	-0.53	1.59	3.10	4.69	3.99
Σ					Stress Check →	0.00

218-83 13-5

WING ANALYSIS

Calculation of Shear Stress

The shear flow at any point on the cross-section is given by:

$$q = (C_1V_x - C_2V_z)(Q_x - Q_{x_0}) + (C_1V_z - C_3V_x)(Q_z - Q_{z_0}) + q_m M_y$$

where

V_x, V_z = components of shearing force along the x-x and z-z axes, respectively

Q_x = ΣA_z, the summation of the product of the element of area and its coordinate from the x-x axis through the centroid of the section

Q_z = ΣA_x $\qquad Q_{x_0} = \dfrac{\Sigma Q_x \Delta a}{\Sigma \Delta a}$ $\qquad Q_{z_0} = \dfrac{\Sigma Q_z \Delta a}{\Sigma \Delta a}$

Δa = Twice the area enclosed by lines joining the ends of an element of skin with the centroid of the entire cross section

q_m = $1/\Sigma \Delta a$ = shear flow for a unit torque

M_y = Torsional moment about the longitudinal axis through centroid of section, positive if clockwise.

Convention of signs for positive direction of the coordinate axes and the shearing forces and moments applied to the cross section are shown below:

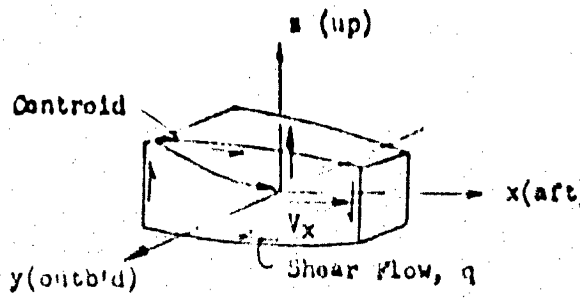
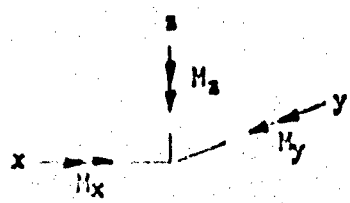


Figure 52a
Port Wing View Looking Inboard



Moments L.H. Rule
Figure 52b

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Shear Flow Factors X_0 & Z_0 Are measd. at beginning of section;

Col	①	②	③	④	⑤	⑥	⑦	⑧	⑨
Item	Ele	Area	x	Ax	z	Az	$Q_1 = \Sigma Ax$	$Q_2 = \Sigma Az$	X_0 Z_0
Ref.				② × ③		② × ⑤	Σ④	Σ⑥	
	32	—	= 23.84	—	= .53	—	0	0	12.64
	1	.85	= 23.84	= 20.26	.53	.45	= 20.76	.45	= 12.96
	2	1.55	= 23.13	= 35.85	1.48	2.29	= 56.11	2.74	= 32.85
	3	1.35	= 21.99	= 29.69	2.20	2.97	= 85.80	5.71	= 44.57
	4	2.40	= 20.27	= 48.65	2.90	6.96	= 134.45	12.67	= 48.18
	5	4.70	= 16.82	= 79.05	3.71	17.44	= 213.50	30.11	= 45.31
	6	4.56	= 12.23	= 55.77	4.28	19.52	= 269.27	49.63	= 12.79
	7	4.61	= 7.64	= 35.22	4.50	20.75	= 304.49	70.38	= 13.73
	8	4.61	3.05	= 14.06	4.47	20.61	= 318.55	93.99	6.93
	9	4.59	1.55	7.11	4.25	19.51	= 311.44	113.50	21.10
	10	4.58	6.14	28.12	3.96	18.14	= 273.32	132.34	40.44
	11	4.62	10.73	49.57	3.58	16.54	= 243.75	145.18	54.85
	12	4.61	15.32	70.63	3.07	14.15	= 163.12	159.33	61.12
	13	4.65	19.91	92.58	2.49	11.58	= 70.54	170.91	56.37
	14	1.04	22.64	23.55	2.04	2.12	= 46.77	173.03	47.55
	15	.62	23.31	14.45	1.57	.97	= 32.54	174.09	37.10
	16	1.40	23.76	33.26	.90	1.26	.72	175.36	21.37
	17	1.40	23.76	33.26	- .90	- 1.26	33.98	174	- 22.08
	18	.62	23.31	14.45	- 1.57	- .97	48.43	173.03	- 35.51
	19	1.04	22.64	23.55	- 2.04	- 2.12	71.98	170.91	- 40.62
	20	4.65	19.91	92.58	- 2.49	- 11.58	164.56	159.33	- 38.15
	21	4.61	15.32	70.63	- 3.07	- 14.15	235.19	145.18	- 32.44
	22	4.62	10.73	49.57	- 3.58	- 16.54	284.76	128.64	- 21.98
	23	4.58	6.14	28.12	- 3.96	- 18.14	317.78	110.50	- 6.14
	24	4.59	1.55	7.11	- 4.25	- 19.51	319.19	90.99	12.46
	25	4.61	- 3.05	- 14.06	- 4.47	- 20.61	305.13	70.38	34.15
	26	4.61	- 7.64	- 35.22	- 4.50	- 20.75	270.71	49.63	55.04
	27	4.56	- 12.23	- 55.77	- 4.28	- 19.52	214.94	30.11	71.79
	28	4.70	- 16.82	- 79.05	- 3.71	- 17.44	135.89	17.67	75.20
	29	2.40	- 20.27	- 48.65	- 2.90	- 6.96	87.24	5.71	63.17
	30	1.35	- 21.99	- 29.69	- 2.20	- 2.97	57.55	2.74	50.34
	31	1.55	- 23.13	- 35.85	- 1.48	- 2.29	21.70	.45	35.78
	32	.85	- 23.84	- 20.26	- .53	- .45	0	0	

PREPARED BY N.C.P.
CHECKED BY
DATE 1-18-61
REVISED

GOODYEAR AIRCRAFT

PAGE 3.01.180
MODEL GA 464
SER- 9801
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...ning of section ; X_e Z_e Are coord. at end of section

WING ANALYSIS

Table XXVII

⑦	⑧	⑨	⑩	⑪	⑫	⑬	⑭	⑮
Q_x = $\Sigma\Delta x$	Q_z = $\Sigma\Delta z$	X_e Z_0	X_0 Z_e	Δa	$Q_x\,\Delta a$	$Q_z\,\Delta a$	$Q_x - Q_{x_0}$	$Q_z - Q_{z_0}$
⑦	Σ②			⑨ - ⑩	⑦ x ⑪	⑧ x ⑪	⑦ - .69	⑧ - 84.50
0	0	12.64	- 13.61	25.28	0	0	- .67	- 84.50
10.16	.45	- 13.96	- 35.31	23.36	- 465.37	11.36	- 10.15	- 84.05
56.11	2.74	- 33.25	- 54.77	18.41	- 1033.6	51.17	- 56.20	- 81.76
85.83	5.71	- 44.57	- 63.77	19.12	- 1645.04	109.52	- 86.47	- 78.79
134.45	12.67	- 47.75	- 75.20	26.42	- 3552.17	334.74	- 135.4	- 71.83
211.59	30.11	- 45.37	- 71.11	26.62	- 5631.57	801.53	- 214.19	- 54.39
257.27	49.63	- 32.73	- 55.04	22.34	- 5315.[illegible]	1108.73	- 267.76	- 34.87
304.49	70.38	- 13.73	- 34.15	20.42	- 6217.61	1437.16	- 305.13	- 14.12
328.55	90.99	6.93	- 12.96	11.27	- 6335.76	1839.79	- 319.24	6.44
311.44	113.50	26.10	6.14	19.96	- 6216.34	2205.58	- 312.13	26.30
283.38	127.64	42.49	21.98	20.51	- 5810.03	2622.41	- 270.01	44.14
235.75	145.18	54.85	32.99	21.91	- 5161.46	3122.17	- 234.44	60.67
164.12	159.33	61.12	38.15	22.97	- 3746.37	3655.41	- 163.41	74.73
70.54	170.91	56.37	40.62	15.75	- 1111.01	2691.23	- 71.93	76.41
46.77	173.03	47.55	35.51	12.01	- 564.35	2078.09	- 47.63	82.43
37.54	174.09	37.30	30.98	15.12	- 331.05	2631.67	- 33.73	84.80
.72	175.36	21.38	- 21.38	42.76	30.79	7498.13	.03	90.76
4.98	174	- 30.98	- 37.30	16.32	550.55	2831.67	33.29	91.50
7.43	173.03	- 35.51	- 47.55	12.01	531.64	2078.09	47.74	82.53
71.96	170.91	- 40.62	- 56.37	15.75	1133.61	2691.23	71.29	86.41
164.56	159.33	- 38.15	- 61.12	22.97	3771.44	3657.41	163.37	74.73
235.19	145.18	- 32.99	- 54.85	21.91	5153.[illegible]	3122.31	234.50	60.63
284.76	127.64	- 21.98	- 42.49	20.51	5840.43	2632.41	270.07	44.14
312.76	110.50	- 6.14	- 26.10	19.96	6245.02	2205.58	312.19	26.03
319.79	90.99	12.96	- 6.93	19.29	6304.60	1701.79	319.30	6.41
305.73	70.38	34.15	13.73	20.42	6247.09	1437.16	305.74	- 14.12
270.71	49.63	55.04	32.70	22.34	6047.66	1108.73	270.02	- 34.27
214.59	30.11	71.99	45.37	26.62	5721.70	801.53	214.25	- 54.39
135.89	12.67	75.20	47.75	26.42	3590.2	334.74	135.20	- 71.83
87.31	5.71	63.77	44.57	19.12	1673.76	101.52	86.55	- 78.79
57.55	2.74	50.39	32.55	18.34	1055.47	50.25	56.86	- 81.74
91.70	.45	35.28	12.26	23.00	499.53	10.36	21.01	- 84.05
0	0							

$$Q_{x_0} = \frac{\Sigma Q_x\,\Delta a}{\Sigma \Delta a} = \frac{470.88}{677.76} = 0.69$$

$$Q_{z_0} = \frac{\Sigma Q_z\,\Delta a}{\Sigma \Delta a} = \frac{57406.88}{677.36} = 84.50$$

$$g_m = \frac{1}{\Sigma \Delta a} = \frac{1}{677.36} = 0.00147$$

2

Σ ⑪ = 677.36
Σ ⑫ = 470.88
Σ ⑬ = 57406.88

Calculation of Shear Stress — WING ANALYSIS

Condition A_2 Symmetrical Maneuver

Wing Sta. 0.00 (Wing Root)

$$q = (C_1V_x - C_2V_z)(Q_x - Q_{x_0}) + (C_1V_z - C_3V_x)(Q_z - Q_{z_0}) + g_m M_y$$

$C_1 = 0$	$V_x = 33$	lbs.	Ref. Pg. 2.02.190
$C_2 = 760.456 \times 10^{-6}$	$V_z = 179$	lbs.	Ref. Pg. 2.02.190
$C_3 = 44.761 \times 10^{-6}$	$M_y = 2070$	In-Lbs	Ref. Pg. 2.02.210
$g_m = 1470 \times 10^{-6}$	$g_m M_y = 3.05$		

$$(C_1V_x - C_2V_z) = -(760.456 \times 10^{-6})(179)$$
$$= -0.136$$

$$(C_1V_z - C_3V_x) = -(44.761 \times 10^{-6})(33)$$
$$= -0.00148$$

$\therefore$

$$q = 3.05 - 0.136(Q_x - Q_{x_0}) - 0.00148(Q_z - Q_{z_0})$$

218-63 (3-5...)

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Condition A2, Wing Sta. 0.00 Table XXVIII Wing Analysis

Col. Element	(14) $Q_A - Q_{A0}$	(15) $Q_F - Q_{F0}$	(16) 0.136 × (15)	(17) 0.00444 × (14)	(18) (16) + (17)	(19) (11) × (18)	Sta.	(20) q in Sta.
1-2	-20.95	-84.05	11.43	0.031	14.51	334	2	14.30
2-3	-56.80	-81.76	11.13	0.084	14.26	261	3	14.08
3-4	-86.49	-78.79	10.71	0.128	13.89	266	4	13.46
4-5	-135.14	-71.83	9.78	0.200	13.03	344	5	11.90
5-6	-214.19	-54.39	7.40	0.317	10.77	287	6	9.49
6-7	-269.96	-34.87	4.75	0.400	8.20	183	7	6.81
7-8	-305.18	-14.12	1.92	0.451	5.42	111	8	4.03
8-9	-319.24	6.49	-0.88	0.473	2.64	53	9	1.31
9-10	-312.13	26.00	-3.54	0.463	-0.03	-1	10	-1.28
10-11	-284.01	44.14	-6.00	0.420	-2.53	-52	11	-3.69
11-12	-234.44	60.68	-8.25	0.347	-4.85	-106	12	-5.88
12-13	-163.81	74.83	-10.19	0.243	-6.90	-158	13	-7.75
13-14	-71.23	86.41	-11.75	0.106	-8.59	-135	14	-8.76
14-15	-47.68	88.53	-12.05	0.071	-8.93	-107	15	-9.02
15-16	-33.23	89.50	-12.20	0.049	-9.10	-149	16	-9.20
16-17	0.03	90.76	-12.35	0.000	-9.30	-399	17	-9.25
17-18	33.23	89.50	-12.20	-0.049	-9.20	-150	18	-9.14
18-19	47.68	88.53	-12.05	-0.071	-9.07	-109	19	-8.94
19-20	71.23	86.41	-11.75	-0.106	-8.81	-139	20	-8.10
20-21	163.81	74.83	-10.19	-0.243	-7.33	-169	21	-6.47
21-22	234.44	60.68	-8.25	-0.347	-5.55	-121	22	-4.46
22-23	284.01	44.14	-6.00	-0.420	-3.37	-69	23	-2.16
23-24	312.13	26.00	-3.54	-0.463	-0.05	-19	24	0.37
24-25	319.24	6.49	-0.88	-0.473	1.69	34	25	3.11
25-26	305.18	-14.12	1.92	-0.451	4.52	92	26	5.96
26-27	269.96	-34.87	4.75	-0.400	7.40	165	27	8.77
27-28	214.19	-54.39	7.40	-0.317	10.13	270	28	11.38
28-29	135.14	-71.83	9.78	-0.200	12.63	334	29	13.13
29-30	86.49	-78.79	10.71	-0.128	13.63	262	30	13.87
30-31	56.80	-81.76	11.13	-0.084	14.10	258	31	14.28
31-32	20.95	-84.05	11.43	-0.031	14.45	332	32	14.50
32-1	-0.69	-84.50	11.50	0.000	14.55	368	1	14.53

Stress Check → $M_y = 2070$ Vs. $\Sigma (19) = 2071$

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<u>Calculation of Shear Stress</u> <u>WING Analysis</u>

Condition C_2 Symmetrical Maneuver

Wing Sta. 0.00 (Wing Root)

$$q = (C_1 V_x - C_2 V_z)(Q_x - Q_{x_0}) + (C_1 V_z - C_3 V_x)(Q_z - Q_{z_0}) + q_m M_y$$

$C_1 = 0$ $\quad V_x = 101$ lbs. Ref. Pg. 7.01.240

$C_2 = 760.456 \times 10^{-6}$ $\quad V_z = 186$ lbs. Ref. Pg. 7.01.110

$C_3 = 44.761 \times 10^{-6}$ $\quad M_y = 2135$ in.-lbs. Ref. Pg. 7.01.260

$q_m = 1470 \times 10^{-6}$ $\quad q_m M_y = 3.14$

$$(C_1 V_x - C_2 V_z) = -(760.456 \times 10^{-6})(186)$$
$$= -0.1416$$

$$(C_1 V_z - C_3 V_x) = -(44.761 \times 10^{-6})(101)$$
$$= -0.00452$$

$$q = 3.14 - 0.1416(Q_x - Q_{x_0}) - 0.00452(Q_z - Q_{z_0})$$

218-63 (3-...)

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Condition C_1, Wing Sta. 0.00 Item XXIX Wing Analysis

Col / Element	(14) $Q_z - Q_{zo}$	(15) $Q_x - Q_{xo}$	(16) -0.1161 (15)	(17) 0.00152 (14)	(18) (16) + (17)	(19) (11) × (18)	Sta	(20) f_0 Sta
1-2	- 20.95	- 84.05	11.90	0.095	15.14	349	2	15.08
2-3	- 56.80	- 81.76	11.58	0.257	14.08	275	3	14.83
3-4	- 86.49	- 78.79	11.15	0.391	14.68	282	4	14.30
4-5	-135.14	- 71.83	10.17	0.610	13.92	368	5	12.87
5-6	-214.19	- 54.39	7.70	0.970	11.81	315	6	10.56
6-7	-269.96	- 34.87	4.94	1.220	9.30	208	7	7.01
7-8	-305.18	- 14.12	2.00	1.380	6.52	133	8	5.09
8-9	-319.24	6.49	- 0.92	1.440	3.66	73	9	2.27
9-10	-312.13	26.00	- 3.68	1.410	0.87	17.4	10	- 0.48
10-11	-284.01	44.14	- 6.25	1.285	- 1.82	- 37.4	11	- 3.11
11-12	-234.44	60.68	- 8.60	1.060	- 4.40	- 96.6	12	- 5.56
12-13	-163.81	74.83	- 10.60	0.740	- 6.72	-154	13	- 7.73
13-14	- 71.23	86.41	- 12.20	0.322	- 8.74	- 138	14	- 8.97
14-15	- 47.68	88.53	- 12.55	0.216	- 9.19	-110	15	- 9.30
15-16	- 33.23	89.50	- 12.70	0.150	- 9.41	-152	16	- 9.56
16-17	0.03	90.76	- 12.05	0.000	- 9.71	- 415	17	- 9.71
17-18	33.23	89.50	- 12.70	- 0.150	- 9.71	-159	18	- 9.67
18-19	47.68	88.53	- 12.55	- 0.216	- 9.63	- 115.5	19	- 9.51
19-20	71.23	86.41	- 12.20	- 0.322	- 9.38	-148	20	- 8.79
20-21	163.81	74.83	- 10.60	- 0.740	- 8.20	-189	21	- 7.36
21-22	234.44	60.68	- 8.60	- 1.060	- 6.52	-143	22	- 5.46
22-23	284.01	44.14	- 6.25	- 1.285	- 4.40	- 90.4	23	- 3.18
23-24	312.13	26.00	- 3.68	- 1.410	- 1.95	- 39	24	- 0.59
24-25	319.24	6.49	- 0.92	- 1.440	0.78	15.5	25	2.27
25-26	305.18	- 14.12	2.00	- 1.380	3.76	77	26	5.31
26-27	269.96	- 34.87	4.94	- 1.220	6.86	153	27	8.37
27-28	214.19	- 54.39	7.70	- 0.970	9.87	263	28	11.29
28-29	135.14	- 71.83	10.17	- 0.610	12.70	336	29	13.30
29-30	86.49	- 78.79	11.15	- 0.391	13.90	267	30	14.18
30-31	56.80	- 81.76	11.58	- 0.257	14.46	265	31	14.71
31-32	20.95	- 84.05	11.90	- 0.095	14.95	345	32	15.02
32-1	- 0.69	- 84.50	11.95	0.000	15.09	382	1	15.12

Stress Check → $M_y = 2135$ vs. Σ (19) $= 2135$

Calculation of Axial Stress

The axial stress (f_a) is simply the axial load (P_y) divided by the perimeter of the wing cross-section (Total of column ①, page 3.01.040).

The axial load (P_y) is taken as the same for conditions A_2 and C_2.

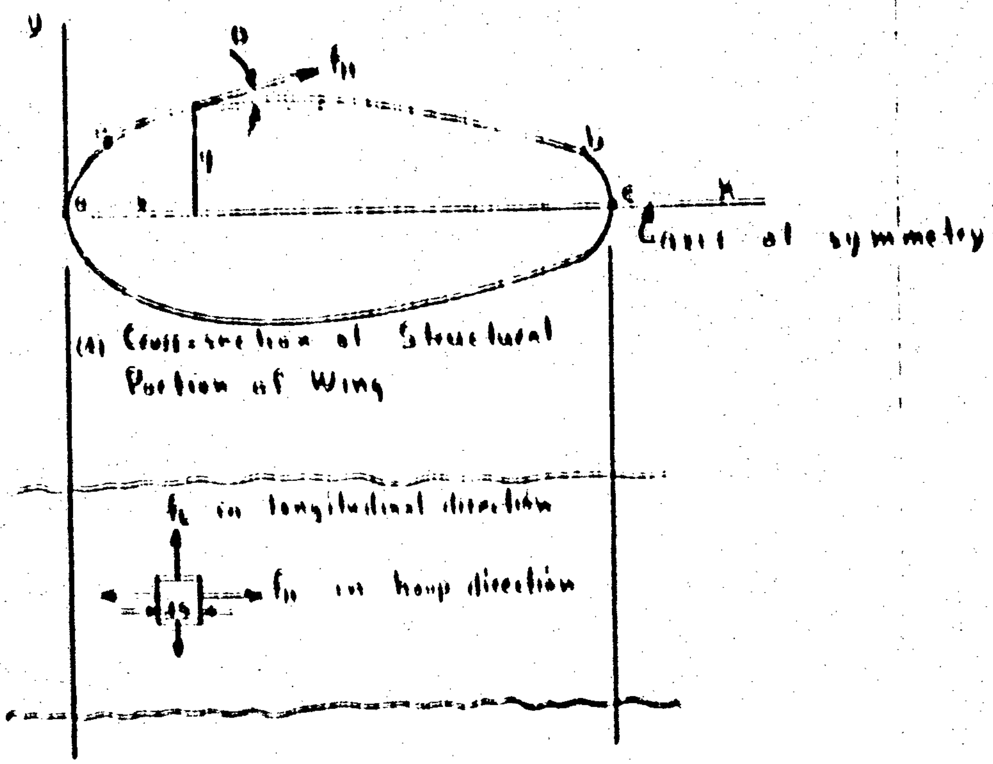
Conditions A_2 and C_2, Wing Sta. 0.00

$P_y = -771$ lbs Ref. Pg. 2.02.170

Perimeter $= 101.48$ in.

$\therefore \quad f_a = -\dfrac{771}{101.48} = -7.56$ lbs./in.

Inflation Stresses in an Airmat Wing — WING ANALYSIS
(see Ref. 3)



(b) Sketch Showing Orientation of Membrane Stresses

Figure 53 Longitudinal and Hoop Inflation Stresses

From summation of forces in x-direction on part of an elemental chordwise strip:

$f_H = \dfrac{py}{\cos\theta}$ = hoop tension, lb/in, y and θ are in (A) and

p = internal pressure, psi. From summation of forces in the longitudinal direction:

$pA = \int_0^c f_L \, ds$, lb. in which A = enclosed

area above x-axis (only half needed because of symmetry),

WING ANALYSIS

f_L = longitudinal stress, lb/in and dS is an element of arc. The longitudinal wall strain e_L is assumed to be constant (i.e. independent of y) and is given by

$$e_L = \frac{1}{E}\{f_L - \mu f_H\}$$ in which E = elastic modulus

of the fabric, lb/in and μ = Poisson's Ratio.

From the above three equations and $dx = ds \cos\theta$

$$f_L = \frac{pA}{S_e} + \frac{\mu p y}{\cos\theta} - \frac{\mu}{S_e}\int_0^{x_e} \frac{y\,dx}{\cos^2\theta}$$

The integral is evaluated in three parts

$$I = \int_0^{x_e} \frac{y\,dx}{\cos^2\theta} = {}_0I_a + {}_aI_b + {}_bI_e$$

${}_0I_a$ and ${}_bI_e$ are for circular arcs $\widehat{0a}$ and $\widehat{be}$ while ${}_aI_b$ is evaluated numerically, using NACA 0015 airfoil coordinates, in the table below. The arc $\widehat{ab}$ is divided into 10 parts referred to as 'points' and $\mu = 0.2$ is assumed.

Table XXX.

Point	y	θ	$\frac{1}{\cos\theta}$	$\frac{y}{\cos^2\theta}$	$\frac{f_H}{p}$	$\frac{f_L}{p}$
	in	Deg			in	in
1	2.56	23.3	1.0888	2.48	2.19	3.130
2	3.71	11.6	1.0197	3.83	3.77	3.460
3	4.18	4.3	1.0028	4.10	4.30	3.566
4	4.50	0	1.0000	4.50	4.50	3.606
5	4.47	0.8	1.0001	4.47	4.47	3.606
6	4.25	3.0	1.0014	4.25	4.25	3.556
7	3.96	4.7	1.0034	3.97	3.78	3.507
8	3.58	5.2	1.0041	3.61	3.55	3.419
9	3.07	6.5	1.0065	3.12	3.09	3.329
10	2.49	6.8	1.0071	2.53	2.51	3.108

The inflation stresses are plotted vs chord positions on the next page. The resultant of f_L does not pass through the centroid of 'A' and hence gives a small chordwise moment that is neglected.

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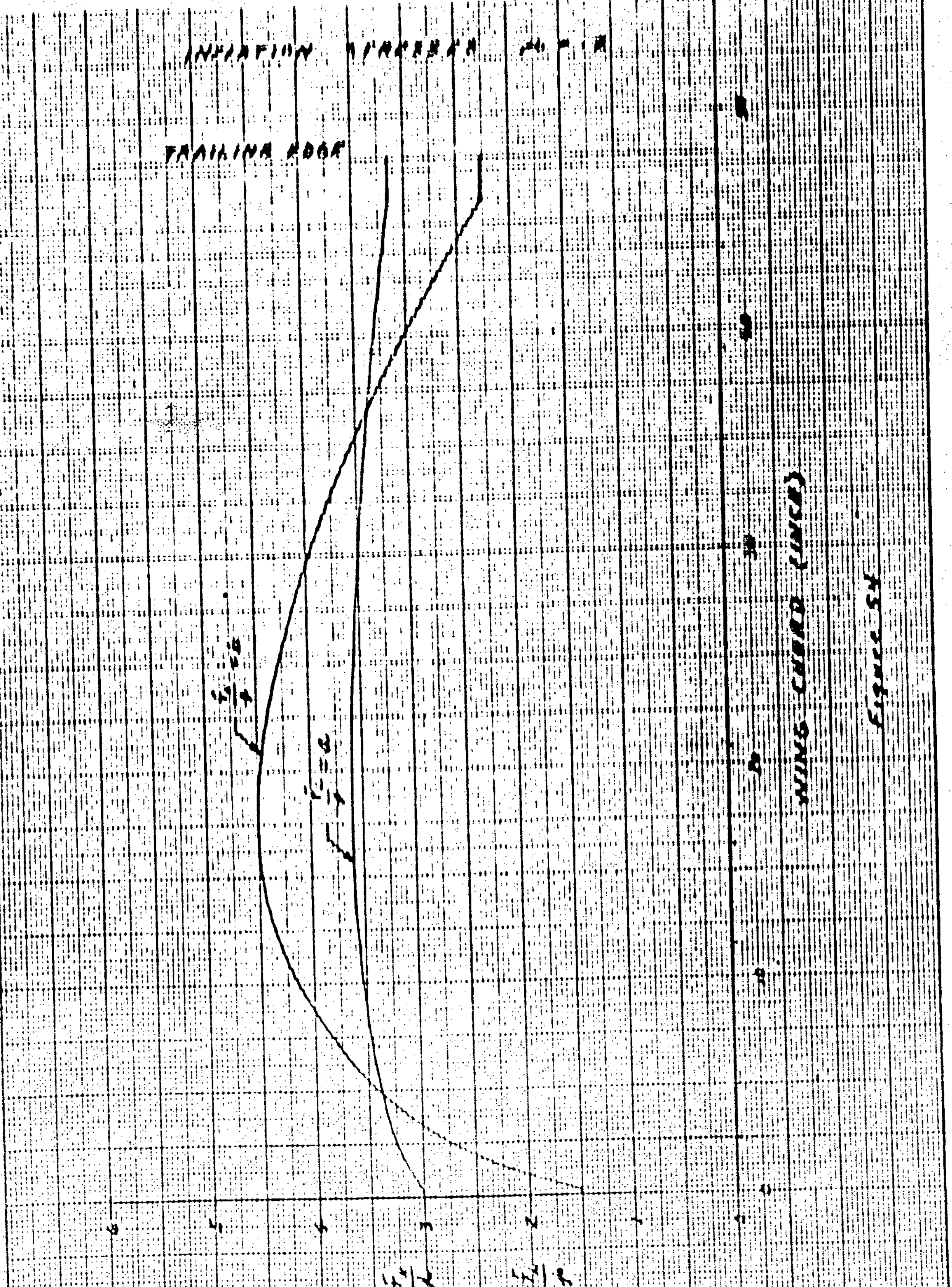


Figure 53

WING ANALYSIS

Calculation of Inflation Stresses

Conditions d_L and C_L, Any Wing Station

These stresses are taken from the figure of page 3.01.180

Table XXXV

Column	①	②
Element Pg. 3.01.040	$a = \frac{f_L}{p}$	$b = \frac{f_H}{p}$
1 & 32	3.00	1.50
2 & 31	3.07	1.92
3 & 30	3.20	2.50
4 & 29	3.33	3.08
5 & 28	3.45	3.70
6 & 27	3.51	4.30
7 & 26	3.60	4.50
8 & 25	3.60	4.45
9 & 24	3.56	4.26
10 & 23	3.50	3.98
11 & 22	3.43	3.60
12 & 21	3.33	3.10
13 & 20	3.22	2.55
14 & 19	3.15	2.27
15 & 18	3.15	2.27
16 & 17	3.15	2.27

Limit Margin of Safety for Wrinkling — WING ANALYSIS

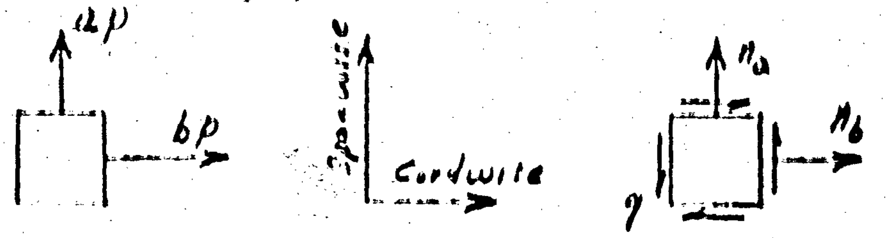
The limit margin of safety for the wing is based on the pressure required to prevent wrinkling of the wing, i.e.

$$M.S. = \frac{P_{inflation}}{P_{req'd}} - 1$$

For $P_{inflation} = 7.0$ psi,

$$M.S. = \frac{7.0}{P_{req'd}} - 1$$

Determination of $P_{req'd}$



Inflation Stresses — Applied Stresses

where:

$$a = \frac{f_L}{p} \qquad b = \frac{f_H}{p}$$

$$N_a = f_a + f_b$$

$$N_b = 0$$

Wrinkling occurs when one of the principal stresses is zero. It may then be seen from a Mohr's stress circle that the shear stress required to cause wrinkling may be expressed as,

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$$q = \sqrt{n_x n_y}$$

where: $n_x = ap + n_a$

$n_y = bp + n_b$

then,

$$q^2 = (ap + n_a)(bp + n_b)$$

$$abp^2 + (bn_a + an_b)p + n_a n_b - q^2 = 0$$

$$p^2 + \left(\frac{n_a}{a} + \frac{n_b}{b}\right)p + \frac{n_a n_b}{ab} - \frac{q^2}{ab} = 0$$

therefore, the pressure required to prevent wrinkling is given by,

$$p = -\frac{1}{2}\left(\frac{n_a}{a} + \frac{n_b}{b}\right) \pm \frac{1}{2}\left[\left(\frac{n_a}{a} + \frac{n_b}{b}\right)^2 + 4\left(\frac{q^2}{ab} - \frac{n_a n_b}{ab}\right)\right]^{\frac{1}{2}}$$

since, $n_b = 0$,

$$p_{Req'd} = -\frac{n_a}{2a} \pm \frac{1}{2}\left[\left(\frac{n_a}{a}\right)^2 + \frac{4q^2}{ab}\right]^{\frac{1}{2}}$$

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Condition A2, Wing Sta. 0.00

Calculation of Margin of Safety for Wrinkling

Col. / Item for A2 / Elem.	(1) f_a 3.01.159	(2) f_b 3.01.003	(3) f_1 = (1) + (2)	(4) q 3.01.188	(5) q^2	(6) a 3.01.170	(7) b 3.01.190	(8) ab
1	-7.56	-2.78	-10.34	14.53	211	3.50	1.50	[illegible]
2	-7.56	-5.19	-12.75	14.23	201	3.09	1.92	5.93
3	-7.56	-6.99	-14.55	14.08	193	[illegible]	2.70	8.00
4	-7.56	-8.71	-16.27	13.40	150.5	3.33	3.38	10.45
5	-7.56	-10.59	-18.15	11.33	141.5	3.45	3.79	13.10
6	-7.56	-11.82	-19.38	9.43	90	3.57	4.19	15.40
7	-7.56	-12.13	-19.69	6.81	46.3	3.62	4.49	16.20
8	-7.56	-11.78	-19.34	4.33	16.2	3.60	4.45	16.00
9	-7.56	-10.91	-18.47	1.31	1.71	3.58	4.26	15.20
10	-7.56	-9.89	-17.45	-1.29	1.64	3.53	3.93	13.90
11	-7.56	-8.66	-16.22	-3.59	13.6	3.43	3.60	12.35
12	-7.56	-7.05	-14.61	-5.58	34.5	3.33	3.10	10.33
13	-7.56	-5.28	-12.84	-7.75	60	3.22	2.55	8.22
14	-7.56	-3.97	-11.53	-8.76	76.5	3.15	2.27	7.15
15	-7.56	-2.70	-10.26	-9.02	81	3.15	2.27	7.15
16	-7.56	-0.94	-8.50	-9.20	84.5	3.15	2.27	7.15
17	-7.56	3.72	-3.84	-9.15	83.3	3.15	2.27	7.15
18	-7.56	5.44	-2.12	-9.17	83.2	3.15	2.27	7.15
19	-7.56	6.63	-0.93	-8.97	80	3.15	2.27	7.15
20	-7.56	7.62	0.06	-5.10	55.5	3.22	2.55	8.22
21	-7.56	8.85	1.29	-6.47	41.8	3.33	3.10	10.33
22	-7.56	9.92	2.36	-4.66	19.3	3.43	3.60	12.35
23	-7.56	10.61	3.05	-2.16	4.66	3.50	3.93	13.90
24	-7.56	11.09	3.53	0.37	0.137	3.56	4.26	15.20
25	-7.56	11.42	3.86	3.11	9.65	3.60	4.45	16.00
26	-7.56	11.23	3.67	5.96	35.5	3.60	4.50	16.20
27	-7.56	10.38	2.82	8.77	77	3.57	4.30	15.40
28	-7.56	8.61	1.05	11.33	129	3.45	3.79	13.10
29	-7.56	6.33	-1.23	13.13	172	3.32	3.08	10.25
30	-7.56	4.41	-3.15	13.57	192	[illegible]	2.70	[illegible]
31	-7.56	2.47	-5.09	[illegible]	203	[illegible]	1.92	[illegible]
32	-7.56	[illegible]	[illegible]	[illegible]	210	[illegible]	[illegible]	[illegible]

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$P_{crit} = -\frac{N_1}{2a} \pm \frac{1}{2}\left[\left(\frac{N_1}{a}\right)^2 + \frac{4\gamma^2}{16}\right]^{\frac{1}{2}}$ WING ANALYSIS

for Wrinkling $M.S. = \frac{7.0}{P_{crit}} - 1$

Table XXXII

(6) a	(7) b	(8) ab	(9) $\frac{4\gamma^2}{16}$	(10) $\frac{N_1}{a}$	(11) $\left(\frac{N_1}{a}\right)^2$	(12) (9) + (11)	(13) $\sqrt{(12)}$	(14) -(10) + (13)	(15) $P_{crit} = \frac{1}{2}$(14)	(16) M.S.
01.140	1.01.140									
3.00	1.50	4.50	187.5	-3.45	11.90	199.40	14.13	17.58	8.79	-0.20
3.07	1.92	5.89	140	-4.15	17.20	157.20	12.54	16.69	8.35	-0.16
3.22	2.73	8.80	99	-4.55	20.70	119.70	10.93	15.48	7.74	-0.10
3.33	3.38	10.45	70.5	-4.90	24.00	94.50	9.74	14.64	7.32	-0.04
3.45	3.79	13.10	43.2	-5.25	27.50	70.70	8.41	13.66	6.83	0.02
3.57	4.33	15.49	23.4	-5.44	29.60	53.00	7.28	12.72	6.36	0.10
3.63	4.73	16.20	11.4	-5.97	35.60	47.00	5.94	11.91	5.96	0.17
3.60	4.45	16.00	4.05	-5.38	28.90	32.95	5.75	11.23	5.62	0.24
3.56	4.26	15.20	0.45	-5.20	27.00	27.45	5.25	10.45	5.23	0.34
3.50	3.98	13.90	0.47	-4.99	24.90	25.47	5.03	10.02	5.01	0.40
3.43	3.60	12.35	4.40	-4.73	22.30	26.70	5.17	9.90	4.95	0.41
3.33	3.10	10.33	13.40	-4.40	19.30	32.70	5.72	10.12	5.06	0.38
3.22	2.55	8.22	29.20	-4.00	16.00	45.20	6.73	10.73	5.37	0.30
3.15	2.27	7.15	42.80	-3.67	13.40	56.20	7.50	11.17	5.58	0.25
3.15	2.27	7.15	45.40	-3.26	10.60	56.00	7.49	10.75	5.38	0.30
3.15	2.27	7.15	46.20	-2.70	7.29	53.49	7.32	10.02	5.01	0.40
3.15	2.27	7.15	47.70	-1.22	1.49	49.19	7.02	8.24	4.12	0.70
3.15	2.27	7.15	46.50	-0.675	0.455	46.96	6.86	7.54	3.77	0.85
3.15	2.27	7.15	44.80	-0.295	0.087	44.89	6.70	7.00	3.50	1.00
3.22	2.55	8.22	31.85	0.019	0.00	31.85	5.65	5.63	2.82	1.48
3.33	3.10	10.33	16.20	0.388	0.15	16.35	4.05	3.66	1.83	2.83
3.43	3.60	12.35	6.41	0.689	0.474	6.88	2.62	1.93	0.97	6.20
3.50	3.98	13.90	1.34	0.872	0.757	2.10	1.45	0.58	0.29	19.41
3.56	4.26	15.20	0.04	0.993	0.985	1.03	1.015	0.33	0.17	40.20
3.60	4.45	16.00	2.41	1.07	1.14	3.55	1.89	0.82	0.41	16.10
3.60	4.50	16.20	8.77	1.02	1.04	9.81	3.12	2.10	1.05	5.66
3.57	4.30	15.40	20.00	0.762	0.58	20.58	4.55	3.89	1.95	2.59
3.45	3.79	13.10	39.40	0.304	0.092	39.49	6.29	5.99	3.00	1.33
3.22	3.03	[illegible]	67.10	-0.37	0.137	67.24	8.21	8.58	4.29	0.63
[illegible]	[illegible]	[illegible]	96.00	[illegible]	0.97	96.97	9.85	10.84	5.42	0.29
[illegible]	[illegible]	[illegible]	137.50	-1.66	2.75	140.25	11.85	13.51	6.76	0.03
[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	6.40	192.90	13.90	16.43	8.22	-0.15

2

Condition C_2, Wing Sta. 0.00

1

Calculation of Margin of Safety for Wrinkling

Col. Item / Arr. Ref. / Elem.	① f_a	② f_b	③ f_c = ① + ②	④ q	⑤ q^2	⑥ a	⑦ b	⑧ ab
Ref.	3.01.153	3.01.183		3.01.173		3.01.193	3.01.193	
1	-7.56	1.51	-6.05	15.12	224	3.00	1.50	4.50
2	-7.56	-1.43	-8.99	15.26?	228?	3.31?	1.92	5.99
3	-7.56	-3.74	-11.30	14.83	220	3.20	2.50	8.00
4	-7.56	-6.06	-13.62	14.30	204	3.33	3.03	10.15
5	-7.56	-8.94	-16.50	12.91?	135?	3.45	3.79	13.10
6	-7.56	-11.25	-18.81	10.56?	111	3.71?	4.39	15.43?
7	-7.56	-12.59	-20.06?	7.91	62.5?	3.69	4.50	16.60?
8	-7.56	-13.81	-21.37?	[illegible]	[illegible]	[illegible]	3.93?	13.00
9	-7.56	-12.95	-20.51	[illegible]	[illegible]	3.56?	4.26?	17.00?
10	-7.56	-12.68	-20.24	-0.48?	0.23	3.53?	4.23?	13.90
11	-7.56	-12.14	-19.70	-3.11	9.65	3.43	3.60	12.35
12	-7.56	-11.20	-18.76	-5.56	30.8	3.33	3.10	10.33
13	-7.56	-10.06?	-17.62	-7.73	59.7	3.22	2.61?	8.41?
14	-7.56	-9.07	-16.63	-8.97	80	3.15	2.27?	7.15
15	-7.56	-7.74	-15.30	-9.30	86.2?	3.15	2.27?	7.15
16	-7.56	-5.79	-13.35	-9.56	91.2	3.15	2.27	7.15
17	-7.56	-0.39	-7.95?	-9.71	94	3.15	2.21?	7.15
18	-7.56	1.68	-5.88	-9.67	93	[illegible]	[illegible]	7.15?
19	-7.56	3.17	-4.39	-9.51	90	3.15?	[illegible]	7.15?
20	-7.56	4.88	-2.68	-8.79?	77	3.22	2.55?	8.22
21	-7.56	7.22	-0.34	-7.36	54?	3.33	3.10	10.33
22	-7.56	9.34	1.78	-5.46	29.8?	3.43	3.60?	12.35?
23	-7.56	11.08?	3.52?	-3.18	10.1	3.50?	3.98	13.90
24	-7.56	12.55	4.99	-0.59?	0.35	3.56?	4.26?	[illegible]
25	-7.56	13.81?	6.25	2.27	5.14	3.60	4.45?	[illegible]
26	-7.56	14.50	6.94	5.31	28.1	3.60	4.50	16.20
27	-7.56	14.43	6.87	8.31	70	3.57	4.39?	15.40?
28	-7.56	13.32	5.76	11.29	127	3.45?	3.79?	13.10?
29	-7.56	11.34	3.78	13.30?	176	3.33?	3.03	10.25
30	-7.56	9.46?	1.90	14.13?	200	3.20?	2.50?	8.00
31	-7.56	7.45	-0.11	14.71	216	3.31?	1.92	5.99
32	-7.56	4.69?	-2.87?	15.0?	225	3.00	1.50	4.50

$$P_{req'd} = -\frac{n_1}{2a} \pm \frac{1}{2}\left[\left(\frac{n_1}{a}\right)^2 + \frac{4q^2}{ab}\right]^{\frac{1}{2}}$$

WING ANALYSIS

ty for Wrinkling $M.S. = \frac{7.0}{P_{req'd}} - 1$

Table XXXIII

6	7	8	9	10	11	12	13	14	15	16
a (3.01.192)	b (3.01.190)	ab	$\frac{4q^2}{ab}$	$\frac{n_1}{a}$	$\left(\frac{n_1}{a}\right)^2$	9 + 11	$\sqrt{12}$	-10 + 13	$P_{req'd} = \frac{1}{2}$ 14	M.S.
3.00	1.50	4.50	202.5	-2.02	4.06	206.6	14.39	16.41	8.21	-0.15
3.31	1.92	5.90	153	-2.93	8.60	161.6	12.73	15.71	7.86	-0.11
3.20	2.50	8.00	110	-3.54	12.50	122.5	11.03	14.63	7.32	-0.04
3.35	3.03	10.65	72.7	-4.10	16.80	89.5	9.46	13.56	6.77	0.03
3.45	3.79	13.10	53.4	-4.79	22.90	73.3	8.57	13.36	6.68	0.05
3.71	4.39	16.30	28.9	-5.27	27.70	56.6	7.52	12.79	6.40	0.09
3.60	4.50	16.20	15.4	-5.59	31.00	46.4	6.81	12.39	6.20	0.13
3.55	3.85	13.60	6.45	-5.70	32.40	38.9	6.24	11.94	5.97	0.17
3.56	3.26	11.60	1.36	-5.75	33.00	34.4	5.87	11.62	5.81	0.20
3.59	3.33	11.90	0.066	-5.85	34.20	34.3	5.86	11.71	5.86	0.19
3.45	3.60	12.35	3.12	-5.75	33.00	36.1	6.02	11.77	5.89	0.17
3.33	3.10	10.33	11.94	-5.64	31.70	43.6	6.61	12.25	6.13	0.14
3.22	2.85	9.17	29.10	-5.48	30.00	59.1	7.70	13.19	6.59	0.06
3.15	2.27	7.15	44.9	-5.39	29.00	72.9	8.54	13.84	6.92	0.01
3.15	2.27	7.15	48.3	-4.85	23.50	71.9	8.49	13.33	6.67	0.05
3.15	2.27	7.15	51	-4.25	18.00	69	8.32	12.57	6.29	0.11
3.15	2.27	7.15	52.5	-2.52	6.34	58.8	7.67	10.19	5.20	0.34
3.15	2.27	7.15	52	-1.87	3.50	55.5	7.45	9.32	4.66	0.50
3.15	2.27	7.15	50.4	-1.40	1.96	52.4	7.25	8.65	4.33	0.62
3.22	2.85	9.17	37.5	-0.84	0.71	38.2	6.19	7.03	3.52	0.99
3.33	3.10	10.33	20.0	-0.10	0.01	20.0	4.57	4.67	2.34	1.99
3.45	3.60	12.[illegible]	3.35	0.52	0.27	9.9	3.15	2.63	1.32	4.30
3.50	3.98	13.90	2.91	1.01	1.02	3.9	1.98	0.97	0.49	13.30
3.[illegible]6	4.2[illegible]	[illegible]	0.032	1.40	1.96	2.1	1.45	0.05	0.03	232.0
3.60	4.1[illegible]	[illegible]	1.29	1.74	3.02	4.3	2.08	0.34	0.17	40.20
3.[illegible]	4.50	16.20	6.95	1.93	3.71	10.7	3.29	1.35	0.68	9.30
3.57	4.30	15.40	18.2	1.93	3.71	21.9	4.69	2.76	1.38	4.07
3.[illegible]	3.79	13.10	38.8	1.67	2.79	41.6	6.46	4.79	2.40	1.92
3.[illegible]	3.05	10.65	68.6	1.14	1.30	69.9	8.36	7.22	3.61	0.94
3.20	2.50	8.00	100	0.59	0.35	100.4	10	9.41	4.71	0.49
3.31	1.92	5.90	146.5	-0.04	0.00	146.5	12.1	12.14	6.07	0.15
3.00	1.50	4.50	200	-0.96	0.92	200.9	14.15	15.11	7.56	-0.08

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WING ANALYSIS

Maximum Stress : Combined Loading

$N_x = a p + N_a = a p + f_a + f_b$

$N_y = b p + N_b = b p + 0 = b p$

Maximum Tensile Stress

$$f_{Max} = \frac{N_x + N_y}{2} \pm \sqrt{\left(\frac{N_x - N_y}{2}\right)^2 + q^2}$$

$$M.S. = \frac{\text{Allowable Limit Strength}}{f_{Max}} - 1$$

Inflation Stress

$$f_{inflation} = b p$$

$$M.S. = \frac{\text{Allowable Inflation Strength}}{b p} - 1$$

Tensile Stress Check

Condition C_2 Element 20 Critical

$$f_{Max} = \frac{24.91 + 26.50}{2} \pm \sqrt{\left(\frac{24.91 - 26.50}{2}\right)^2 + 11.29^2} = 37.60 \text{ \#/in.}$$

$$M.S._{Limit} = \frac{174 \div 3}{37.60} - 1 = +0.46$$

Ref. Pg. 1.00.080 For 174 #/in Allowable

Inflation Stress Check

Element 26 Critical

$b p = 4.50 \times 7 = 31.50$ #/in.

$$M.S._{inflation} = \frac{174 \div 4}{31.50} - 1 = +0.39$$

Static Test And Wind Tunnel M.S. Comparison

Ref. Pg. 1.00.050 For g's.

Static Test M.S.

$$M.S._{ult.} = \frac{5.60}{2.5 \times 1.75} - 1 = +0.28$$

Wind Tunnel M.S.

$$M.S._{ult} = \frac{5.13}{2.5 \times 1.75} - 1 = +0.17$$

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FUSELAGE ANALYSIS

Section 4

FUSELAGE ANALYSIS

The fuselage is an inflated conical envelope with nearly hemispherical ends. It supports external loads by virtue of tensile inflation stresses. If one of the principal stresses at a point in the fuselage becomes zero due to applied compression stresses, a wrinkle starts to form there. As more load is applied the wrinkle will enlarge to a point where collapse will occur. The pressure in the fuselage should be large enough to prevent a wrinkle from forming under limit loads and collapse at ultimate loads. For this analysis the ultimate load is 1.75 times limit load.

While the minimum principal stress at the point where a wrinkle first forms is zero, both principal stresses at a diametrically opposite point are tensile stresses. The material properties should be such that the maximum tensile stress does not exceed some allowable value. The allowable strength value on the tension side is the quick break value derived from a cylinder burst test divided by a creep rupture factor of 3. The factor of 3 accounts for the fact that fabric under load for a period of time has a reduction in strength.

The allowable hoop tension strength value on the compression side is the quick break value derived from the cylinder burst test times 0.65 divided by 3. The factor of 0.65 accounts for the fact that the fabric in a cylinder burst test is loaded at a 2-1 stress ratio, while on the compression side of the fuselage is loaded in only one direction and consequently gets little or no help from the bias ply. The factor is based on a comparison of cylinder burst tests and strip tensile tests.

MEMBRANE STRESSES IN A CONE — FUSELAGE ANALYSIS

INFLATION STRESSES

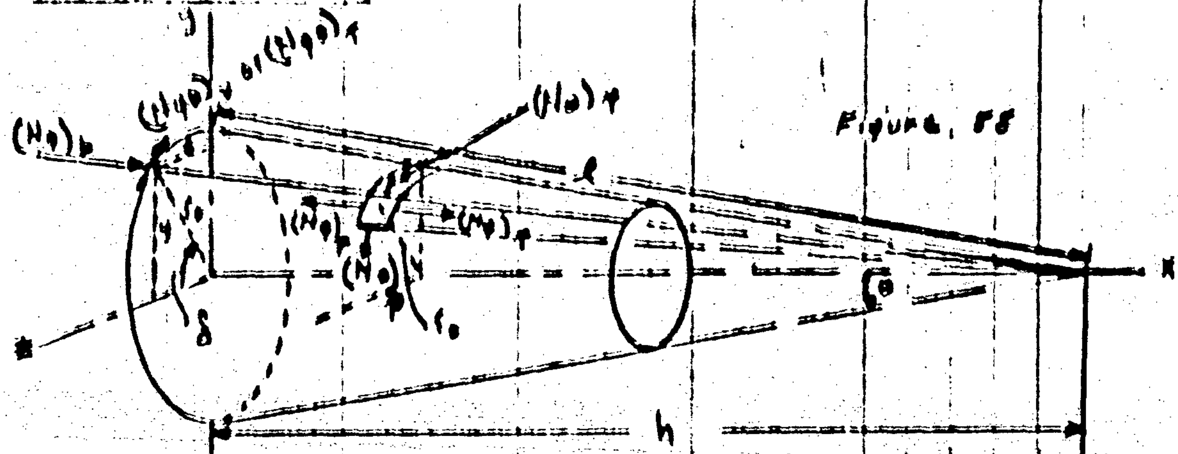


FIGURE 58

THE INFLATION STRESSES $(N_\phi)_p$ AND $(N_\theta)_p$ ARE GIVEN IN REF. 6 PAGE 437.

$$(N_\phi)_p = \frac{p r_0}{2\cos\theta}, \text{ LB/IN} = \text{LONGITUDINAL INFLATION STRESS}$$

$$(N_\theta)_p = \frac{p r_0}{\cos\theta}, \text{ LB/IN} = \text{HOOP TENSION}$$

WHERE p = INFLATION PRESSURE, PSI.
r_0 = RADIUS OF NORMAL SECTION, IN.
θ = ONE HALF OF ANGLE OF CONE.

BENDING STRESSES

IF THE BENDING STRESS $(N_\phi)_b$ IS ASSUMED TO ACT ALONG A GENERATOR OF THE CONE THEN ITS X-COMPONENT MUST SATISFY THE FLEXURE FORMULA

$$N_x = \frac{h}{\ell} N_\phi = \frac{M_z y}{I}$$ WHERE M_z IS THE MOMENT ABOUT THE Z-AXIS.

OR $$(N_\phi)_b = \frac{M_z y}{I}\frac{\ell}{h} = \frac{M_z y}{I\cos\theta}$$

THE MAXIMUM VALUE OF $(N_\phi)_b$ IS

$$(N_\phi)_{bm} = \frac{M_z}{\pi r_0^2 \cos\theta}$$

BEAM SHEAR STRESSES.

FUSELAGE ANALYSIS

IF THE NET SHEAR AT A SECTION IS ALONG THE Y-AXIS, THEN THE SHEAR STRESS IS GIVEN BY THE ELEMENTARY SHEAR STRESS FORMULA TO BE

$$(N_{\phi\theta})_V = \frac{V \cos\theta}{\pi r_0} \quad \text{LB/IN}$$

TORSIONAL SHEAR STRESSES

THE TORSION OF THIN WALLED SHELLS GIVES

$$(N_{\phi\theta})_T = \frac{T}{2\pi r_0^2} \quad \text{LB/IN}$$

218-83 (3-5...)

1

Table XXIV

(1) Condition and Section Ref. Pg. 2.03.200 & 2.03.210	(2)	(3) Location on Circum.	(4) M	(5) V	(6) T	(7) Th (Thrust)	(8) ro	(9) $(N_\theta)_p$ $\frac{T(7)}{\cos\theta}$	(10) (N_θ) $\frac{1}{2}(9)$
			IN-LB	LB	IN-LB	LB	IN		
F-2	SECTION AT POINT O	a / b	15480	124	0	-30	12.28	86.0	43.
F4	SECTION AT POINT O	a / b	12700	94	432	-30	12.28	86.0	43.
F11 & F13	SECTION BETWEEN 7 & 8	a' / b'	4520	111	5270	-58	8.62	60.3	30.
F11 & F13	SECTION AT POINT O	a' / b'	15400	77	5270	-58	12.28	86.0	43.0

F-2 OR F4 AT SECTION AT POINT O

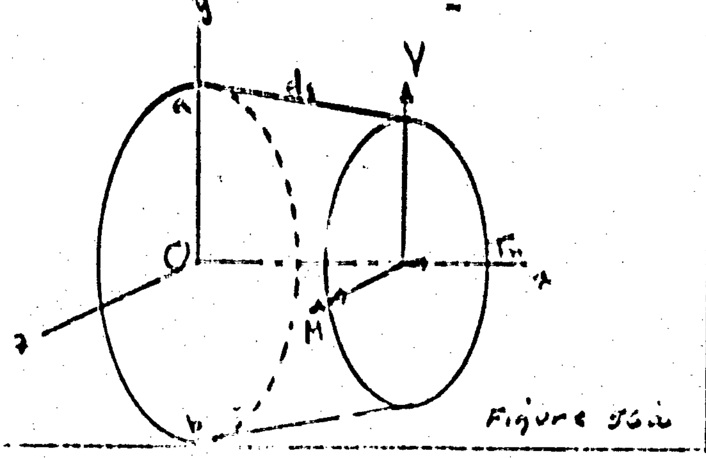
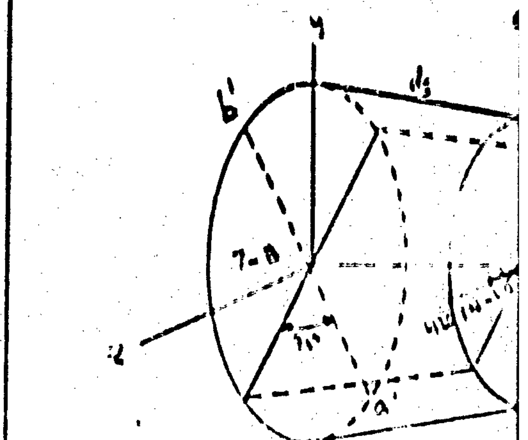


Figure 55.

F11 & F13 SECTION BETWE



$$P_{req} = -\frac{1}{2}\left(\frac{n_a}{a} + \frac{n_b}{b}\right) + \frac{1}{2}\sqrt{\left(\frac{n_a}{a} + \frac{n_b}{b}\right)^2 + 4\left(\frac{q^2}{ab} - \frac{n_a n_b}{ab}\right)}$$

$\therefore P_{req} = \frac{4.76}{2}$

$= 5.8$

$a \cdot p = (10)$ F11 & F13 AT SECTION BETWEEN POINTS 7 & 8

$b \cdot p = (9)$ $\frac{n_a}{a} = -\frac{19.4 + 1.1}{4.31} = -4.76$

$M.S. = \frac{7}{5.85}$

$n_a = (11) + (12)$

FROM A MOHR'S

$n_b = 0$ $\frac{q^2}{ab} = \frac{\overline{15.4}^2}{4.31 \times 8.62} = 6.46$

$N_{max} = \frac{N_\theta +}{2}$

$q = (17)$

NOTE: IN COLUMN (17) IT IS CONSERVATIVE TO ADD (13) TO (14)

(8)	(9)	(10)	(11)	(12)	(13)	(14)	(15)	(16)	(17)	(18)	(19)
r_0	$(N_\theta)_p$ $\frac{2(11)}{\cos\theta}$	$(N_\phi)_p$ $\frac{1}{2}(9)$	$(N_\phi)_{bin}$ $\frac{(11)}{(8)^2 \pi \cos\theta}$	$(N_\phi)_c$ $\frac{(7)}{(8)\, 2\pi}$	$(N_{\phi\theta})_V$ $\frac{(5)}{(8)\pi}$	$(N_{\phi\theta})_T$ $\frac{(6)}{(8)^2\, 2\pi}$	N_θ (9)	N_ϕ (10)+(11)+(12)	$N_{\phi\theta}$ (13)+(14) or (13)	N_{max} Principal Stress	P_{req}
IN	LB/IN										PSI
12.28	86.0	43.0	±32.9 +32.9	=.4	3.2	0	86.0	9.7 75.5	3.2	86.0	5.42
13.38	86.0	43.0	±26.9 +26.9	=.4	2.4	.5	86.0	13.9 61.5	2.4	86.4	4.49
8.64	60.3	30.1	±19.4 +19.4	=1.1	4.1	11.3	60.3	9.6 47.4	15.4	70.9	5.75
12.28	86.0	43.0	±32.7 +32.7	−.8	1.9	5.6	86.0	9.5 74.9	7.5	89.8	5.60

F13 SECTION BETWEEN POINTS 7 & 8 ---------- SECTION AT POINT O

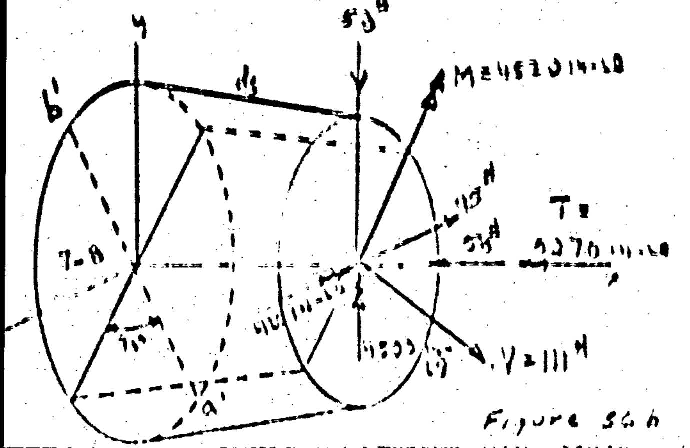


Figure 56b

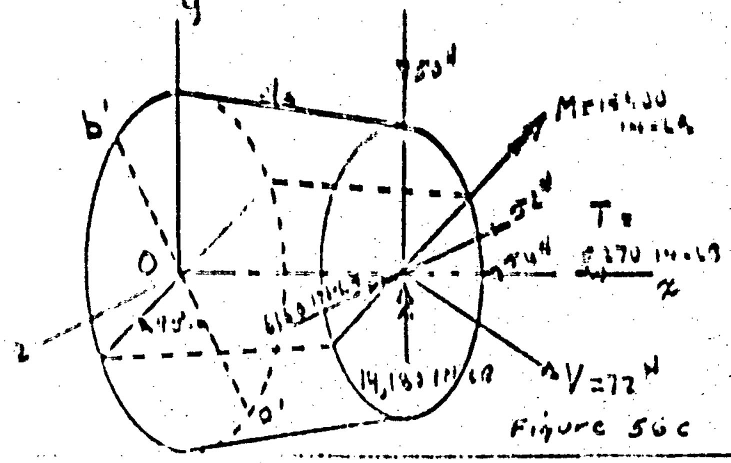


Figure 56c

$$\therefore P_{req} = \frac{4.76}{2} + \frac{1}{2}\sqrt{4.76^2 + 4(6.46)}$$

$$= 5.85 \text{ psi}$$

$$M.S. = \frac{7}{5.85} - 1 = .20$$

FROM A MOHR'S CIRCLE

$$N_{max} = \frac{N_\theta + N_\phi}{2} + \sqrt{\left(\frac{N_\theta - N_\phi}{2}\right)^2 + N_{\phi\theta}^2}$$

FOR F11 & F13 AT SECTION AT POINT O

$$N_{max} = \frac{86 + 74.9}{2} + \sqrt{\left(\frac{86 - 74.9}{2}\right)^2 + 7.5^2}$$

N_{max} = 89.8 LB/IN

Inflation stress at max. radius

$N_{max.} = 7(13.5) = 94.5$ $\quad M.S. = \frac{410}{94.5 \times 4} - 1 = +0.08$

HOOP STRESS MARGIN

$$M.S. = \frac{410 \times .65}{86 \times 3} - 1 = +.03$$

ENGINE MOUNT ANALYSIS

Section 5

ENGINE MOUNT ANALYSIS

An engine saddle mount is attached to the top of the wing by lacing to patch attachments on the top surface of the wing. The saddle mount is also used to fasten the wing to the fuselage, at the trailing edge, by means of two short cable-patch attachments. The engine is attached to the pylon of the saddle structure through the four engine mount fittings. The mount is a weldment of 6061-T6 aluminum tubing heat treated after welding.

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Summary Loads and Stresses of Engine Mount

Table XXXI

Member and Loading Condition		Section Properties: A (in^2)	I_x (in^4)	I_y (in^4)	Limit Loads: M_x (in-lb)	M_y (in-lb)	Torque (in-lb)	Axial (lb)	Ultimate Stress: f_b (psi)	f_s (psi)	f_{comb} (psi)	Allowables: F_b (psi)	F_s (psi)	F_{comb} (psi)	Margin of Safety
-3A Support [illegible] Oval -55 Gusset	$n_z=-1$ Flight	.3701	.0257	.3690	478	3306	315	-2722	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	.315
-41 Strut [illegible] Streamline	Side Load	.3457	.0057	[illegible]	[illegible]	[illegible]	[illegible]	220	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
-5 [illegible] and -17, -35; [illegible] x .049 Tubes together	$n_z=-1$ Flight	.609		[illegible]		[illegible]	705		[illegible]	[illegible]		[illegible]	[illegible]		[illegible]
-53 Tubes [illegible]	$n_z=-1$ Flight	.1464	[illegible]	.0166				[illegible]			[illegible]			[illegible]	[illegible]
-65 Tubes 1" x .049"	$n_z=-1$ Flight	.1464	.0166	.0166				-2154			[illegible]			[illegible]	[illegible]
-29 Tube Ring 1" x .065"	$n_z=+3.0$	[illegible]	[illegible]	[illegible]	1361	[illegible]	1048		[illegible]	[illegible]		[illegible]	[illegible]		.22
-57 Tube 3/4" x .049"	$n_z=-3$	.1079	.0067	.0067	[illegible]	405	405		[illegible]	[illegible]		[illegible]	[illegible]		[illegible]
-5 Tube 1" x .049"	Side Load	.1464	.0166	.0166	707	8	352		21,200	7850		[illegible]	[illegible]		[illegible]

NOTE: Changes in some members have not been accounted for in the section properties as larger margins would result.

Ref. Dwg. 447A-011. For loads see pg. 2.04.000.

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SUMMARY OF

EMPENNAGE, COCKPIT, AND LANDING GEAR ANALYSES

Section 6

Summary of Empennage, Cockpit, and Landing Gear

Description of Empennage

The horizontal tail is supported along its center line on the fuselage and by cables attached to outboard edges. The main support cables are the aft cables which are attached near the hinge line of the elevator while forward cables are added for stability. Thus half of the horizontal stabilizer acts as a cantilever beam supported at the end.

The vertical tail is hinged to the horizontal tail and to the fuselage and also supported by cables; the main cable being attached near the rudder hinge line and a forward cable added for stability is attached to the leading edge. As the largest tail loads are applied to the vertical tail and as the vertical tail is not supported as well as the horizontal tail, the vertical tail is the most critically loaded part of the empennage.

Description of Cockpit

The cockpit is made up of flat sections of Airmat, three inches thick, consisting of two side panels, a bottom panel, a seat bulkhead, and a rear bulkhead. These panels are joined to form the cockpit which attaches to the fuselage section. A hammock type seat is provided for the pilot, which is cemented to the top of the rear bulkhead and seat bulkhead.

Description of Landing Gear

A single wheel landing gear is used on the aircraft. The structure supporting the wheel is a weldment of 6061-T6 aluminum tubing heat treated after welding, and is attached by lacing to patch attachments on the forward end of the fuselage. This structure is designed to provide a shock absorber action during landing. The wing lower brace cables are attached to the channel support structure at the rear of the landing structure.

Stress Analysis of Empennage

The empennage is an Airmat structure and supports the external loads by making use of the tensile inflation stress. The pressure in the empennage should be large enough to prevent collapsing due to the applied compression stresses under ultimate loads. On the tension side the allowable strength value is the quick break strength derived from a cylinder burst divided by the reduction factors shown on page 1.00.110.

STRESS ANALYSIS OF EMPENNAGE

FROM PG 9.05.000 THE UNIT REACTION GIVES

$$\left.\begin{array}{ll} \text{SHEAR} & V = 111 \text{ LB} \\ \text{MOMENT} & M = 245 \text{ IN-LB} \\ \text{TORQUE} & T = 444 \text{ IN-LB} \\ \text{SECTION WIDTH} & = 33.4 \text{ IN} \end{array}\right\} \text{FOR THE CRITICAL SECTION}$$

FROM PG 9.05.000 THE CRITICAL LOAD, CORRECTED FOR VERTICAL TAIL INERTIA, IS 161 LB; HENCE, THE UNIT VALUES ABOVE BECOME;

CRITICAL CONDITION A II RUDDER KICK

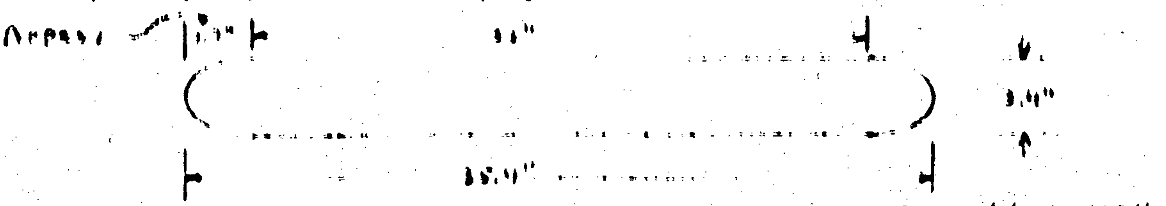
$V = 1.61 \times 11.4 = 18.5$ LB

$M = 1.61 \times 245 = 397$ IN-LB

$T = 1.61 \times 477 = 772$ IN-LB

$P_c = 1.61 \times 33.4 = 40.4$ LB

THE SECTION PROPERTIES AT THE CRITICAL SECTION ARE:



REF DWG 44?? -004

$I = \frac{\pi}{2}(1.7)^3 + 32(1.7)^2(4) = 185.9 \text{ in}^3$

$A = \pi(1.7)^2 + 33(3.4) = 117.9 \text{ in}^2$

$S = 3.4\pi + (2)(33) = 74.7 \text{ in}$

THE APPLIED STRESSES ARE;

$n_x = \frac{Mc}{I} + \frac{P_c}{S} = \frac{397(1.7)}{185.9} + \frac{40.5}{74.7} = 4.17 \text{ LB/IN}$

$q = \frac{T}{2A} = \frac{772}{2(117.9)} = 3.28 \text{ LB/IN}$

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EMPENNAGE ANALYSIS

THE INFLATION STRESSES ARE

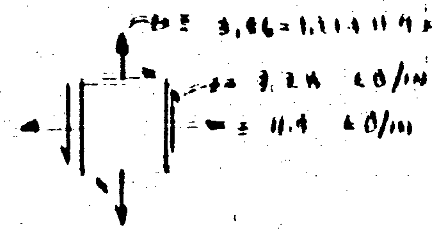
$$a\,p = b\,p = 1.7(7) = 11.9 \text{ LB/IN}$$

$$P_{1,2} = \tfrac{1}{2}\left(\frac{n_a}{a} + \frac{n_b}{b}\right) \pm \tfrac{1}{2}\sqrt{\left(\frac{n_a}{a} - \frac{n_b}{b}\right)^2 + 4\left(\frac{S^2}{ab} - \frac{n_a n_b}{ab}\right)}$$

$$= \tfrac{1}{2}\left(\frac{4.33}{1.7}\right) \pm \tfrac{1}{2}\sqrt{\left(\frac{4.33}{1.7}\right)^2 + 4\left(\frac{3.28}{1.7}\right)^2} = 3.80 \text{ PSI}$$

$$M.S. = \frac{7}{3.80} - 1 = .84$$

THE MAXIMUM FABRIC STRESSES ARE (ON OPPOSITE SIDE)



$$f_{max} = \frac{14.35 + 11.9}{2} + \sqrt{\left(\frac{14.35 - 11.9}{2}\right)^2 + (3.28)^2} = 16.60 \text{ LB/IN}$$

THE FABRIC FACTORS OF SAFETY ARE

REF PG. 1.00.080

$$M.S. = \frac{140}{4 \times 11.9} - 1 = 1.96 \quad \text{ON INFLATION STRESS}$$

$$M.S. = \frac{140}{3 \times 16.6} - 1 = 1.81 \quad \text{ON MAXIMUM LIMIT STRESSES}$$

$$M.S. = \frac{140}{1.5 \times 16.6 \times 1.75} - 1 = 2.20 \quad \text{ON MAXIMUM ULT. STRESSES}$$

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COCKPIT STRESS ANALYSIS — REF DWG 447A-008

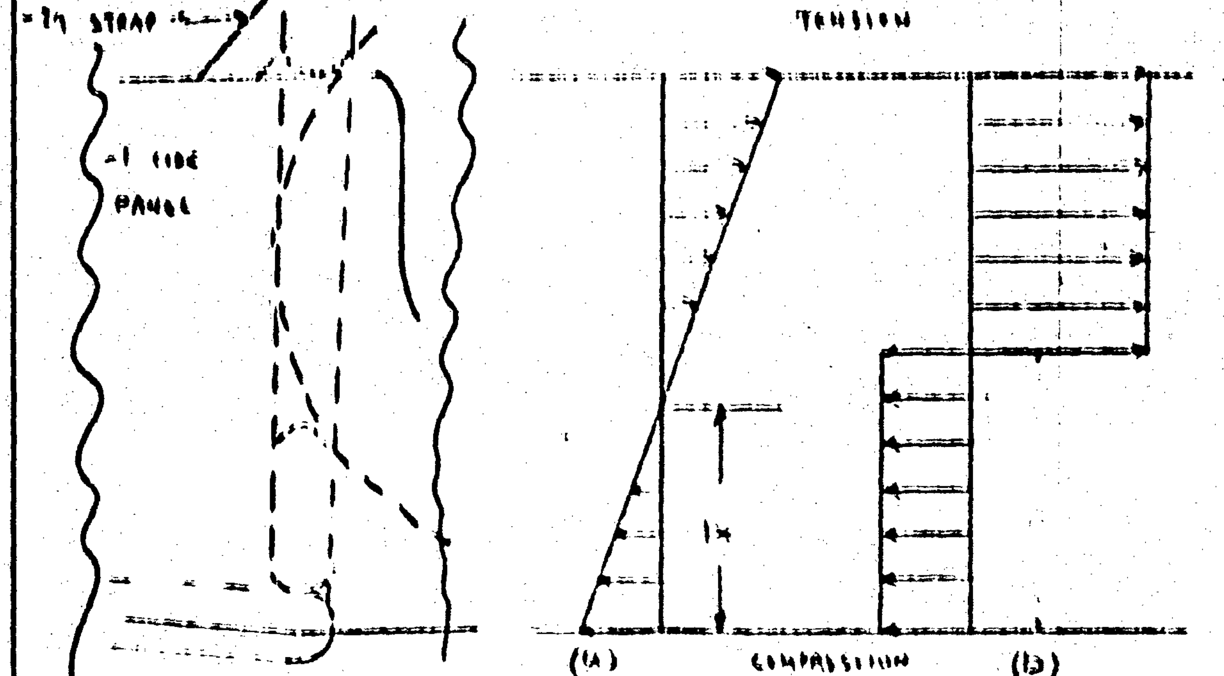


Figure 57

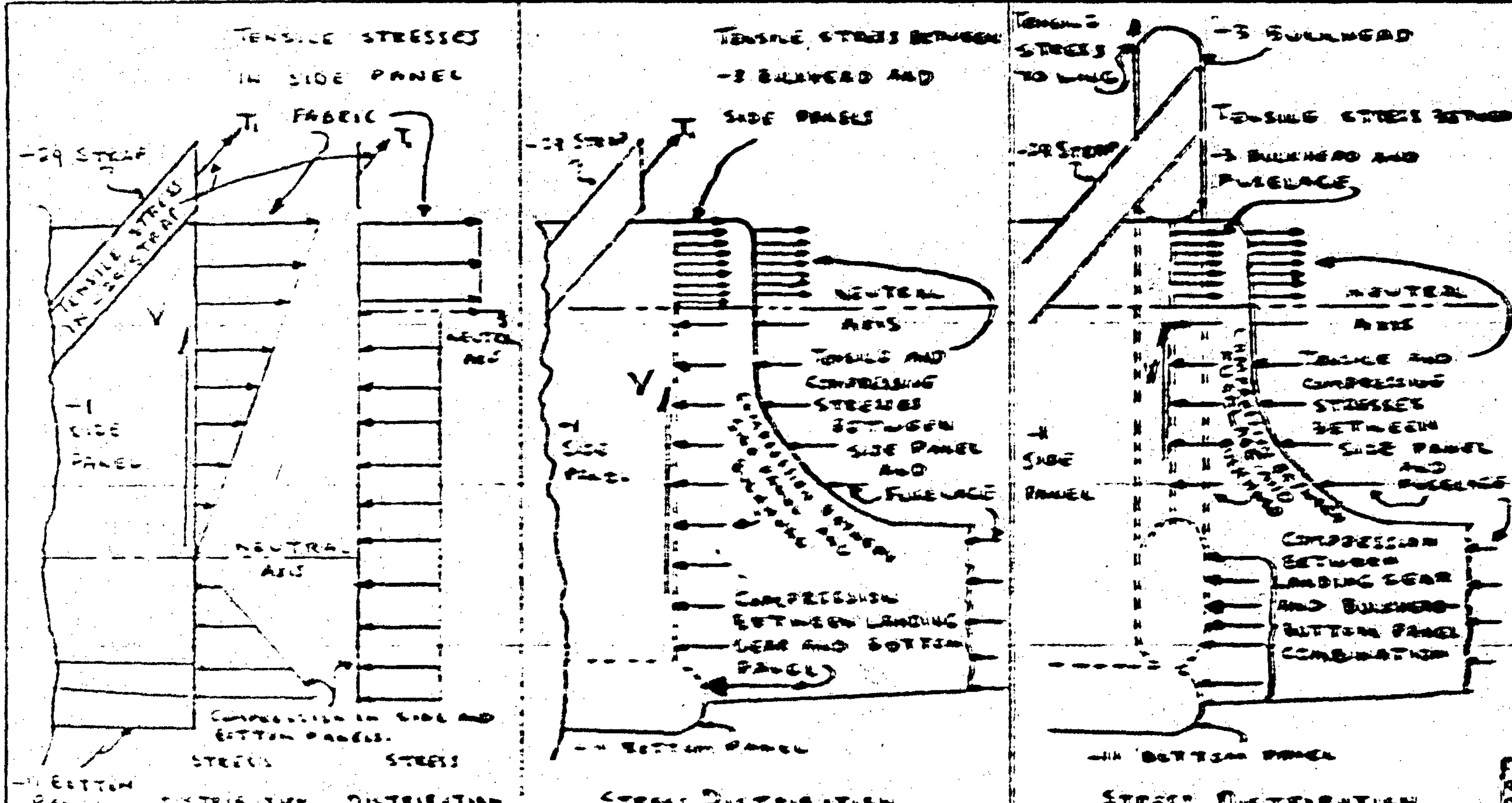
(A) STRESS DISTRIBUTION UP TO WRINKLING OF BOTTOM PANEL

(B) STRESS DISTRIBUTION UP TO COLLAPSE OF COCKPIT

THE ORDINARY BEAM BENDING STRESS DISTRIBUTION IS ASSUMED UP UNTIL THE BENDING STRESSES IN THE BOTTOM PANEL EQUAL THE INFLATION STRESSES. AFTER THE BOTTOM PANEL WRINKLES IT CANNOT CARRY ANY ADDITIONAL COMPRESSIVE LOAD, SO THE SIDE PANELS BEGIN TO WRINKLE PROGRESSIVELY UPWARD. THIS SHIFTS THE NEUTRAL AXIS UP AS SHOWN IN THE SKETCH ABOVE. NEAR THE COLLAPSE LOAD OF THE COCKPIT IT IS ASSUMED THAT THE TENSILE STRESS DISTRIBUTION ABOVE THE NEUTRAL AXIS IS UNIFORM AS SHOWN IN FIGURE (b) ABOVE.

ON THE NEXT PAGE ARE MORE DETAILED SKETCHES SHOWING THE TRANSFER OF LOADS FROM THE COCKPIT SIDE AND BOTTOM PANELS TO THE BULKHEAD, WING, LANDING GEAR, AND FUSELAGE. AS LONG AS THE COCKPIT DOES NOT COLLAPSE, THE LOAD CAPABILITIES ARE DEPENDENT UPON THE FABRIC TENSILE STRENGTH AND THE PRESSURE.

COCKPIT ANALYSIS



STRESS DISTRIBUTION (a)

STRESS DISTRIBUTION (b)

STRESS DISTRIBUTION (c)

STRESS DISTRIBUTION (d)

THE FREE-BODY CONSISTS OF THE COCKPIT JUST FORWARD OF THE -3 BULKHEAD.

THE FREE-BODY CONSISTS OF THE COCKPIT SIDE PANELS, BOTTOM PANEL, THE -29 STRAP; BUT NOT THE BULKHEAD

THE FREE-BODY CONSISTS OF THE COCKPIT SIDE PANELS, BOTTOM PANEL, AND BULKHEAD. THE -29 STRAP FORCES ARE INTERNAL.

Figure 5E

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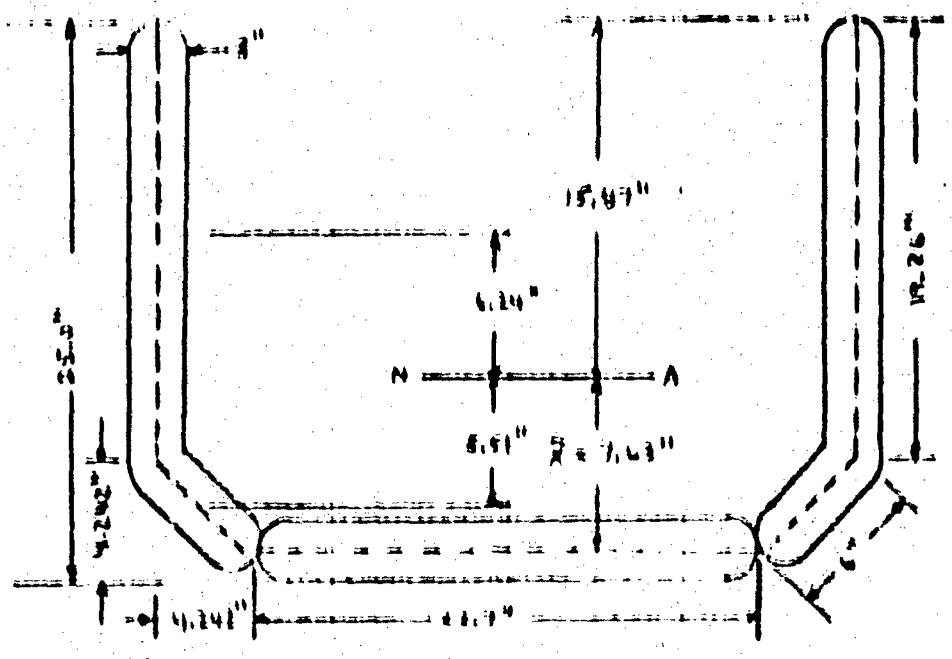
STRESS DISTRIBUTION (A) COCKPIT ANALYSIS



SECTION JUST FORWARD OF BULKHEAD

Figure 54

SECTION PROPERTIES USING CENTERLINE DIMENSIONS

$$\left[(19.26+6)2 + 12.7\right]\bar{y} = 19.26\left(4.24 + \frac{19.26}{2}\right)2 + 6(2.12)(2)$$

$$\bar{y} = 7.63"$$

$$I_{NA} = \left[\frac{19.26^3}{12} + 19.26(6.24)^2 + \frac{6^3}{12}(.707)^2 + 6(5.51)^2\right]2 + 12.7(7.63)^2$$

$$= 4345 \text{ in}^4$$

SECTION MODULUS FOR BOTTOM PANEL = $\frac{4345}{7.63}$ = 577 in^3

" " " TOP OF SIDE PANEL = $\frac{4345}{15.87}$ = 277 in^3

WRINKLING STRESS OF BOTTOM PANEL = (3)(1)(7) = 21 #/in.

WRINKLING MOMENT = (21)(577) = 12,120 IN-LB.

THIS NEGLECTS THE CONTRIBUTION T_j OF THE -19 STRAP

218-83 (3-55)M

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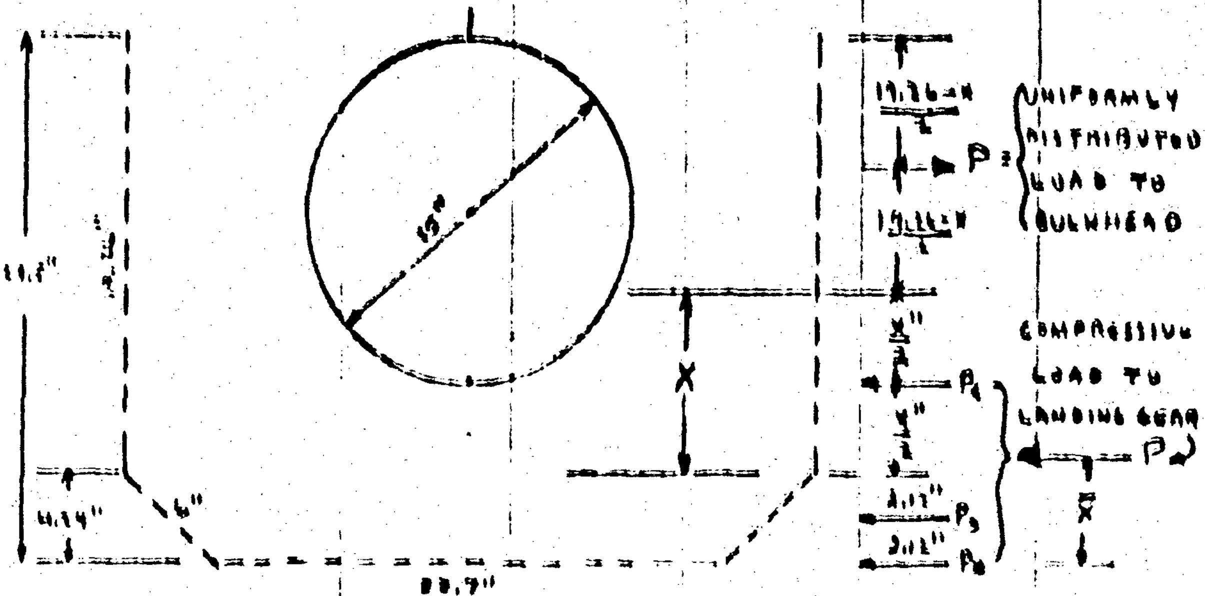
STRESS DISTRIBUTIONS (A) AND (C) — CHECKOFF ANALYSIS



LET p = INFLATION PRESSURE — Figure 60

$P_B = 3(22.7)p = 68.1p$, LB = WRINKLING LOAD OF BOTTOM PANEL

$P_s = 2[3(6)]p = 36.0p$, LB = " " " SLOPING PORTION

$P_x = 2[3(X)]p = 6Xp$, LB = " " " X" OF SIDE PANEL

$P = (68.1 + 36.0 + 6X)p = (104.1 + 6X)p$ = COMPRESSIVE LOAD

$$\bar{X} = \frac{2.12(36p) + (4.24 + \frac{X}{2})(P_x)}{P} = \frac{76.36 + 25.45X + 3X^2}{104.1 + 6X}$$

$$M = \left(23.5 - \frac{19.26 - X}{2} - \bar{X}\right)P$$

or

$$\frac{M}{p} = 1368 + 109.8X$$

THE UNIFORMLY DISTRIBUTED TENSION IS $T_u = \frac{P}{(19.26 - X)(2)}$ LB/SIDE PANEL

OR $\frac{T_u}{p} = \frac{104.1 + 6X}{2(19.26 - X)}$ LB/IN OF SIDE PANEL

THESE SAME EQUATIONS DESCRIBE STRESS DISTRIBUTION (C) AS LONG AS THE FORCES $P, P_x, P_s, \& P_B$ ARE THE FORCES IN THE CONNECTIONS

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STRESS DISTRIBUTION (d) CANOPY ANALYSIS

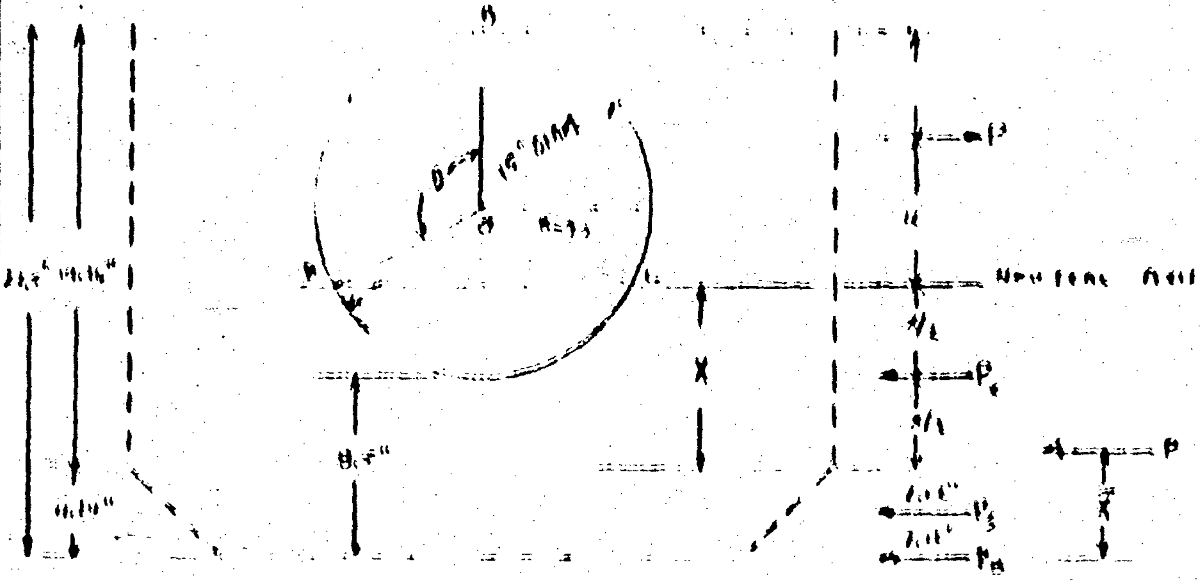

Figure A1

$P_8 = 144\,p$, LB
$P_6 = 36.0\,p$, LB
$P_5 = 64\,p$, LB

These forces are now the compressive forces between fuselage and bulkhead, side panel and fuselage, and bulkhead and landing gear. Their numerical values are not actually needed to evaluate the remaining load of the bottom and side panels.

The upper tensile load P is actually distributed along the arc $\widehat{ABC}$ and between the side panels and fuselage. The portion along the arc $\widehat{ABC}$ also contains the effect of the strap load T_1. In the computations below all of the tensile load is assumed to go along the arc $\widehat{ABC}$.

u = distance to centroid of $\widehat{ABC}$ $= 7.5\left(\frac{\sin\beta}{\beta} - \cos\beta\right)$

$x = 8.5 + 7.5 - 4.24 - 7.5\sin(90-\beta) = 11.76 - 7.5\cos\beta$

$\bar{x}$ = same value as for (b) & (c)

$M = (u + x + 4.24 - \bar{x})\,P$

$P = (104.1 + 6x)\,p$ as before

$T_4 = \frac{P}{15\beta}$ LB/IN

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COCKPIT ANALYSIS

SUMMARY OF EQUATIONS AND SAMPLE CALCULATIONS

STRESS DISTRIBUTION (a)

SEE PG. 6.01.060

CENTERLINE STRESS = $\frac{MC}{I} = S$ LB/IN

FABRIC STRESS = $\frac{S}{2}$, LB/IN

INFLATION STRESS = $\frac{p \times \text{PANEL THICK}}{2}$

$= 1.5p$ LB/IN

NET FABRIC TENSION = $\frac{S}{2} + 1.5p$ LB/IN

THE LIMIT MOMENT FOR CONDITION I PG 2.06.030 IS 11,125 IN-LB

$S = \frac{11,125 \times 15.67}{4345} = 40.2$ LB/IN

NET TENSION IN COCKPIT FABRIC $\Big\}$ $= \frac{40.2}{2} + 1.5(7)$

$= 30.6$ LB/IN.

TENSION IN STRAPS BETWEEN BULKHEAD AND SIDE PANELS

$= 40.2$ LB/IN

STRESS DISTRIBUTIONS (b) AND (c)

SEE PG 6.01.070

$P = (104.1 + 6x)p$ LB

$\bar{x} = \frac{76.36 + 35.45x + 3x^2}{104.1 + 6x}$ IN.

$M = (368 + 10.5x)p$ IN-LB

$T_u = \frac{P}{2(4.2 - x)}$ LB/IN.

= STRESS IN CONNECTIONS, I.E. STRESSES (d) ARE CARRIED BETWEEN BULKHEAD AND FUSELAGE CONNECTIONS

FABRIC TENSION IN SIDE PANEL $\Big\} = \frac{T_u}{2}$ LB/IN

NET FABRIC TENSION $\Big\} = \frac{T_u}{2} + 1.5p$

FOR $x = 10"$, $p = 7$ PSI

$P = 1149$ LB

$\bar{x} = 3.85$ IN.

$M = 17,260$ IN-LB

$T_u = 62.0$ LB/IN

$\frac{T_u}{2} = 31.0$ LB/IN

NET TENSION $= 31 + 10.5 = 41.5$ LB/IN

STRESS DISTRIBUTION (d)

SEE PG. 6.01.070

P AND $\bar{x}$ AS IN (b) AND (c)

$u = 7.5\left(\frac{\sin\beta}{\beta} - \cos\beta\right)$

$x = 11.76 + 7.5\cos\beta$

$M = (u + x + 4.2 - \bar{x})P$

$T_u = \frac{P}{15\beta}$

FOR $x = 10"$, $p = 7$ PSI

$P = 1149$ LB

$\bar{x} = 3.85"$

$\cos\beta = -.2346$

$\beta = 180 - 76.4 = 103.6°$

$= \frac{103.6}{57.3} = 1.81$ RADIANS

$u = 5.80"$

$M = 15,600$ IN-LB

$T_u = 42.3$ LB/IN

THE EXPRESSIONS FOR M AND T_u ARE PLOTTED VS. X ON THE NEXT PAGE

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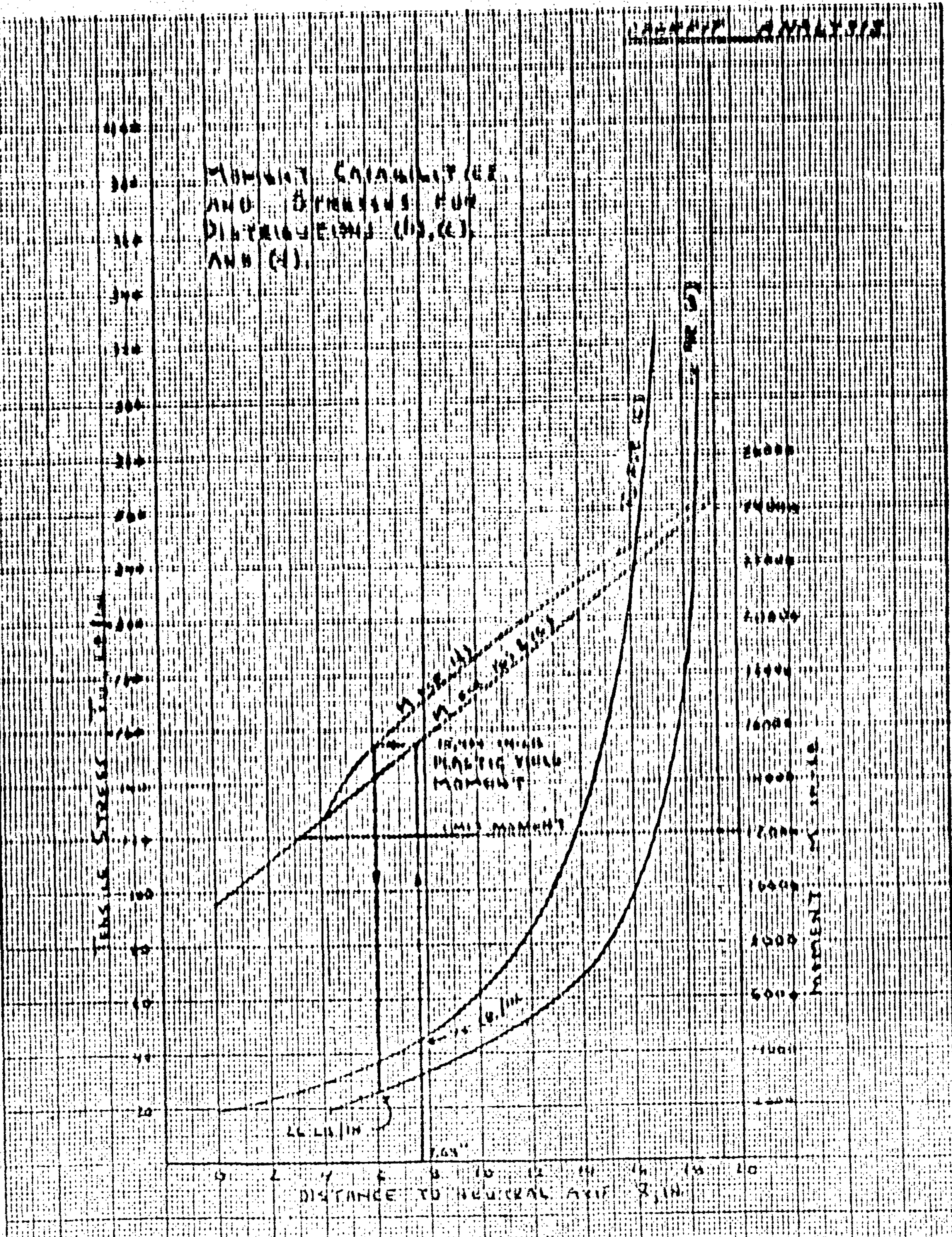

Figure 62

COCKPIT ANALYSIS

From page 6.01.080 the neutral axis for triangular stress distribution (a) is at $\bar{x} = 7.63$ in. The curves on page 6.01.130 give the moments and stresses vs. neutral axis location for plastic yield or uniform stress distributions (b), (c), and (d). Changing from stresses (a) to (b) or (c) at $\bar{x} = 7.63$ changes the limit moment 12,120 in-lb (page 6.01.120 to an initial plastic yield moment of 15400 in-lb (page 6.01.120). Thus:

For $M \leq 12120$ in-lb - - - - - - - - stresses (a)
For $12120 \leq M \leq 15400$ in-lb - - - transition
For $M \geq 15400$ - - - - - - - - - - - - stresses (b) and (c)
} stresses (d)

The curves of page 6.01.140 are derived from those of page 6.01.120 by simultaneous values of M and T_u for $15400 \leq M \leq 23000$. The stresses $(T_u)_b$ are calculated from

$$(T_u)_b = \frac{1}{2}(T_u)_c + \frac{3}{2}(7) = \frac{1}{2}(T_u)_c + 10.5 \text{ lb/in.}$$

in which the stress 10.5 lb/in is the cockpit panels inflation stresses.

The fabric factors of safety are three on limit stress, 1.5 on stress at ultimate load, and four on inflation stress. The cockpit panels are made from Airmat fabric A351 with cylinder burst values of 150 x 150 lb/in warp and fill, the bulkhead is fastened to the side and bottom panels with N313A105 fabric straps with strip tensiles of 200 x 100 lb/in warp and fill; and the bulkhead and side panels are fastened to the fuselage with ZX-300 fabric straps with 425 x 350 lb/in warp and fill strip tensiles.

The limit and ultimate moments are given in the table below.

Condition	1	5	6
Limit Moment	11,125	15,773	13,916
Ultimate Moment	19,440*	20,800**	18,400**

* 11,125 x 1.75 = 19,440 In-lb - - - - Using a factor of safety 1.75
** 15,773$\sqrt{1.75}$ = 20,800 In-lb - - - - Using a factor of safety of 1.75 on energy of absorption for landing, and noting that loads are proportional to $\sqrt{\text{energy}}$.

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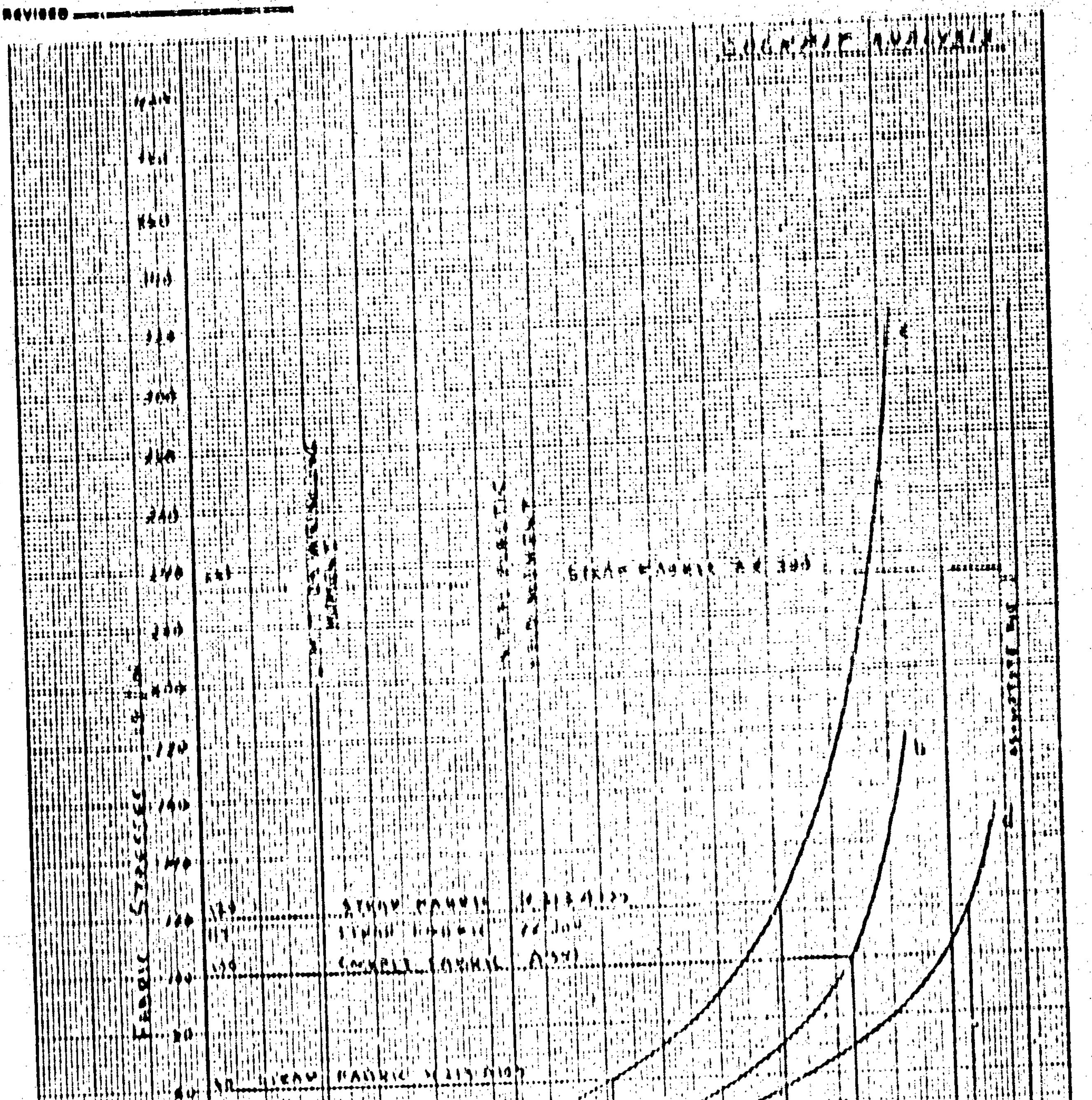

Figure 63

COCKPIT ANALYSIS

TABLE XXXVI

FABRIC ALLOWABLE STRESSES AND MOMENTS

	F_{tu}	$\frac{F_{tu}}{1.5}$	M_{ult}	$\frac{F_{tu}}{3}$	M_{lim}	Initial Plastic Moment	Wrinkling Moment
	LB/IN	LB/IN	IN-LB	LB/IN	IN-LB	IN-LB	IN-LB
Cockpit Fabric A 351	150	100	21300	50	15400	15400	12120
Strap Fabric N313 A105	180	120	24100	60	17103	15400	—
Strap Fabric EX 300	350	233	>23000	117	>23000	—	—

FABRIC MARGINS OF SAFETY

	On Ultimate Moment			On Limit Moment			On Initial Plastic Moment		
Condition	1	5	•	1	5	6	1	5	6
Cockpit Fabric A 351	.1	.07	.16	.56	.17	.32	.32	-.02	-.1
Strap Fabric N313 A105	.03	-.02	.09	.54	.07	.22	.32	-.02	-.1
Strap Fabric EX 300	>.18	>.1	>.25	>1.87	>.46	>.65	—	—	—

Condition	M.S. on Wrinkling
1	.09
5	-.23
6	-.13

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LANDING GEAR ANALYSIS

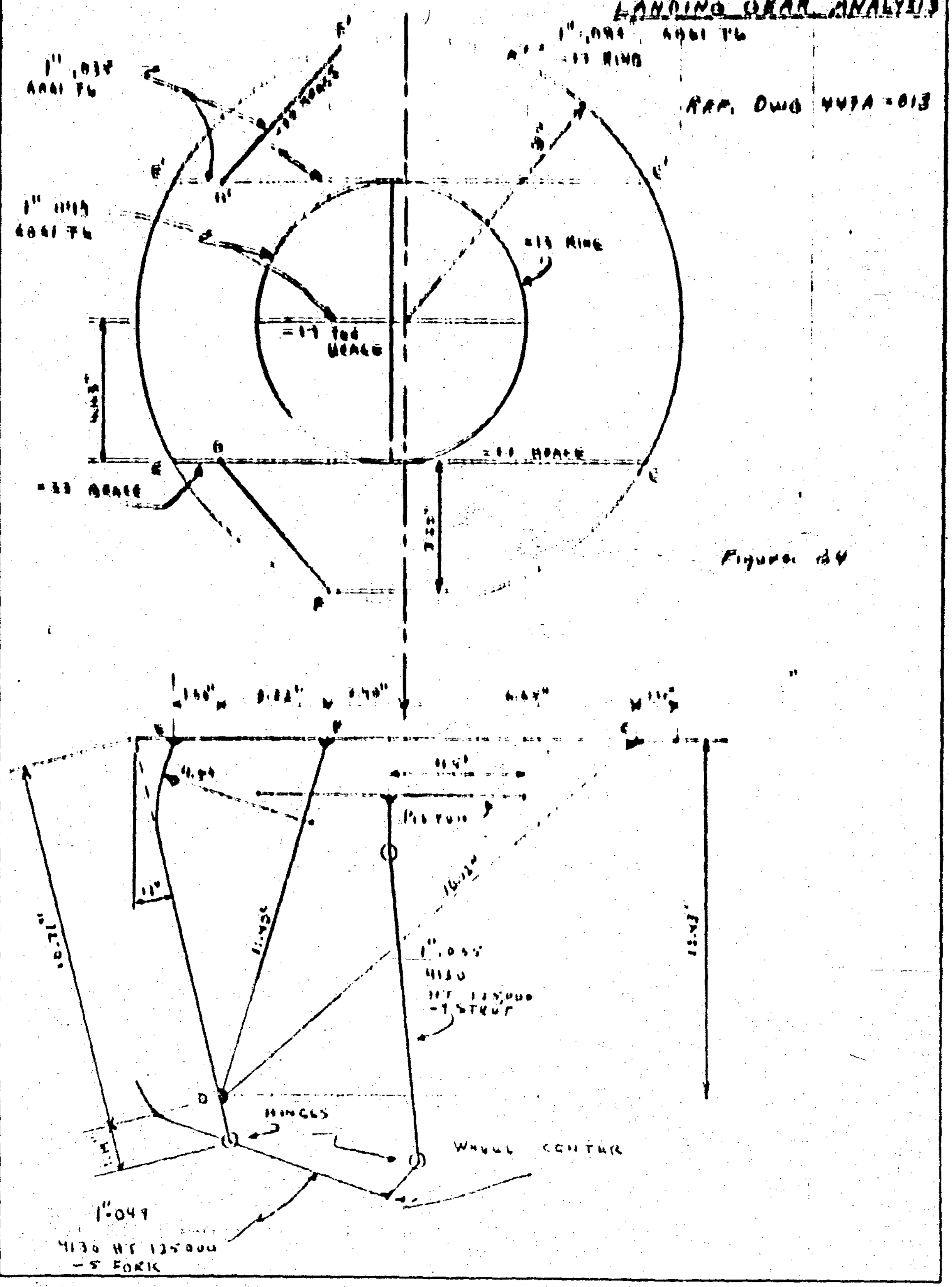

Figure 84

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LANDING GEAR ANALYSIS

STRESS ANALYSIS: SUMMARY OF LANDING GEAR - GAC DWG 447A-813

Table XXXVII

MEMBER	SIZE AND MATERIAL	CONDITION	ULTIMATE LOADS			ALLOWABLES			MARGIN OF SAFETY
			THRUST	MOMENT	TORQUE	F_{tu}	F_b	F_s or F_{st}	
			LB	IN-LB	IN-LB	PSI	PSI	PSI	
-23 FWD BRACE	1" .035" 6061-T6	(3)	-537	537	0	42,000	48,700		.64
-30 MIDDLE BRACE		(3)	-1470	0	0	42,000	—		AMPLE
-27 AFT BRACE		(3)	+1480	0	0	42,000	—		AMPLE
-13 RING AND -17 TEE BRACE		(3)	0	4460	0	42,000	51,200		.14
-37 RING	1" .063" 6061-T6	(3)	0	1905	1249	42,000	64,500	26,000	.41
-4 STRUT	1" .035" SAE 4130	(1) & (2)	-990	0	0	75,000	—	108,000	HIGH
-5 FORK	1" .065" SAE 4130	(4)	+1353	3152	0	125,000	152,000	—	.32